DETERMINATION OF LONGITUDINAL STABILITY PARAMETERS BY STEADY STATE FLIGHT TESTING AND THEORETICAL CALCULATIONS FOR THE RYAN NAVION

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DETERMINATION OF LONGITUDINAL STABILITY PARAMETERS BY STEADY STATE FLIGHT TESTING AND THEORETICAL CALCULATIONS FOR THE RYAN NAVION AIRPLANE

Summary

A series of steady state flight tests were made to determine the longitudinal stability parameters for the Ryan Navion Airplane, and the results were compared to values calculated from theoretical considerations.

It was found that the steady state flight test values for the stability parameters agreed very closely with the theoretical values. The differences averaged approximately 6% except for the damping derivatives $C_{m d\theta}$ and $C_{m d\alpha}$ in which greater differences were encountered. These differences were not consistent, in that in some cases the power-on values agreed closely while in other conditions the power-off values were more nearly equal. Similarly, the clean and landing conditions had no trend in the differences between the steady state flight test values and the theoretical values, with the result that assignment of error to either analysis is not feasible.

Introduction

Previous investigations have determined the longitudinal airplane stability parameters from flight tests for numerous aircraft models and configurations using dynamic or static flight testing methods. The results of these investigations have been compared with those determined from theoretical calculations. It is the purpose of this investigation to afford a basis of comparison for all three methods of parameter determination.

This report may be considered as part I and II of a three part investigation. Parameters were determined by steady state flight testing and by theoretical considerations. Report No. 231 determines the longitudinal airplane stability derivatives by dynamic testing methods. A comparison of the parameters obtained by all three methods are attached to this report as an appendix.

The steady state flight testing was done at four center of gravity positions using conventional flight procedures and instrumentation. Two configurations, landing and clean, were tested under power-on and power-off conditions.

This investigation was conducted at Princeton University during the spring term of 1953.

PART I

THEORETICAL STABILITY ANALYSIS

Procedure and Discussion

The theoretical treatment of the various stability derivatives and airplane parameters was based primarily on methods outlined in Reference (1) and supported by various NACA reports, and other textbooks. The flight conditions that were selected corresponded to actual conditions encountered in the steady state flight tests. That is, for clean power-on condition, the airspeed and power required was as determined from actual flight tests.

It was assumed throughout the theoretical development that the airplane behavior was essential linear and that certain higher order effects were small and could be neglected.

The equation for the summation of moments about the airplane center of gravity can be written:

$$C_{mcg} = C_{L_w} \frac{Xa}{C} + C_{mac} + C_{mFUS} - C_{Lt} \overline{V} \eta_t + T_c \frac{2D^2}{S_w} \frac{z}{c}$$

$$+ C_{np} \frac{l_p}{S_w} \frac{S_p}{} = 0$$

Since the tail lift coefficient is the only unknown, it can be found by substituting the following relation for C_{L_w}.

$$C_{L_w} = C_{La} - C_{Lt} \frac{S_t}{S_w} \eta_t$$

Having solved for C_{Lt}, it is now possible to solve for the derivative $\frac{dC_m}{dC_L}$ from the following relation, which is simply the derivative of the moment equation:

$$\frac{dC_m}{dC_L} = \frac{Xa}{C} + \frac{dT_c}{dC_L}\,\frac{2D^2}{S_w}\,\frac{z}{C} + \frac{dC_{np}}{dC_L}\,\frac{l_p}{S_w}\,\frac{S_p}{C}$$

$$- \frac{a_t}{a_w}\,\bar{V}\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right)\eta_t - C_{Lt}\,\bar{V}\,\frac{d\left(\frac{vs}{v}\right)^2}{dC_L} + \left(\frac{dCm}{dC_L}\right)_{FUS.}$$

The determination of the various derivatives are as shown in "Sample Calculations," and the values of the various constants are as shown in the sections concerning airplane data, notation, and specifications.

The control fixed neutral point can now be evaluated, since $N_o = X_{cg} - \frac{dCm}{dC_L}$.

In order to determine the elevator hinge parameters C_{h_α} and C_{h_δ}, it was necessary to use wind tunnel data contained in Reference (7). The two dimensional data was then corrected to three dimensional flow and further correcte for overhang balance by methods contained in References (1) and (8).

The elevator power, $C_{m_\delta} = - a_t\,\bar{V}\,\eta_t\,\tau$, as developed in Reference (1), can be computed from known values. The factor, τ, can be obtained from a plot in the above reference and is based on the ratio of elevator area to horizontal tail area.

The Navian airplane has a mass unbalance in the elevator control system which produces a nose down moment. This unbalance was measured and found to be equal to 10 ft. lbs.

It can be shown (Reference (1) that:

$$N_o - N_o' = -\frac{C_{m_\delta}\,C_{h_\alpha}}{a_w\,C_{h_\delta}}\left(1 - \frac{d\epsilon}{d\alpha}\right) - \frac{HM_b\,C_{m_\delta}}{W/S\;C_{h_\delta}\;S_e\;c_e}$$

where: HM_b = mass unbalance = 10 ft. lbs.

From the above relation N_o' can be computed since all factors are known or can be found.

Similarly, in steady turning flight:

$$N_m = N_o - \frac{63\ g\ l_t \rho C_{m_\delta}}{2\ W/S} \left(1 + \frac{1}{n^2} \right)$$

and,

$$N_m' = N_o' + \frac{57.3\ g\ l_t \rho C_{m_\delta}}{2\ W/S\ C_{h_\delta}} \left(1 + \frac{1}{n^2} \right) \left(C_{h_\alpha} - \frac{1.1\ C_{h_\delta}}{\tau} \right)$$

where n = normal acceleration.

Two other derivatives of interest can be computed from known values, namely:

$$C_{md\alpha} = - a_t \bar{V} \eta_t \frac{1}{\mu} \frac{l_t}{C} \frac{d\epsilon}{d\alpha}$$

and

$$C_{md\theta} = - 1.1\ a_t \bar{V} \eta_t \frac{l_t}{C} \frac{1}{\mu}$$

$C_{md\theta}$ is a damping derivative and accounts for the tail damping in pitch. The factor 1.1 was used to account for the damping of the rest of the airplane which was assumed to be 10% the value of the tail damping.

The derivative $C_{m\alpha}$ was computed in two ways in order to compare with steady state and with dynamic flight test results. For steady state comparison:

$$C_{m\alpha} = C_{L\alpha} (X_{cg} - N_o)$$

For dynamic comparison:

$$C_{m\alpha} = C_{L\alpha} (X_{cg} - N_m)$$

In order to determine the slope of the airplane lift curve, power off, it can be assumed that:

$$C_{La} = C_{L_w} + C_{Lt}$$

It can then be shown that:

$$C_{L_\alpha} = \frac{a_w + a_t \frac{S_t}{S_w} \eta_t \left(1 - \frac{d\epsilon}{d\alpha}\right)}{1 - \left(\frac{dCm}{dC_L}\right)_{FIX} \frac{c}{l_t}}$$

where $\left(\frac{dCm}{dC_L}\right)_{FIX} = X_{cg} - N_o$

The foregoing indicates the methods used to determine the various derivatives and parameters theoretically. In the "Sample Calculations" section, the complete solution of one flight condition is shown.

Where applicable, an altitude of 6,500 ft. was assumed which was the average altitude encountered in flight tests. In maneuvering flight calculations, an average value of acceleration was used which from flight tests was found to $n = 1.3$.

Navion Specifications and Notations:

Wing area, S_w = 184.2 ft^2

Horizontal Tail area, = S_t = 43 ft^2

Elevator area = 15.04 ft^2

Aspect Ratio, Wing = A = 6.04

Aspect Ratio, Tail = A_t = 4.02

Wing Span = b = 33.38 ft.

Tail Span = b_t = 13.17 ft.

MAC Wing, C = 5.7

MAC Tail = C_t = 3.34

Tail length, l_t = 15.04 ft.

Incidence, Wing, i_w = 2° (at root), = 1° (at tip)

Incidence, Tail, i_t = -3°

Airfoil, Wing, = Root:NACA 4415 R, Tip:NACA 6410 R

Airfoil, Tail, = NACA 0010

Dihedral = 7.5°

Wing Taper Ratio = λ = .54

Wing Aerodynamic Center, a. c. = .242

Horizontal Tail, Taper Ratio = λ_t = .67

Length overall = 27.25 ft.

Wing Root Chord = 7.2 ft.

Wing Tip Chord = 3.92 ft.

Elevator:

Root Chord = 1.5 ft.

Tip Chord = 1.0 ft.

Stabilizer:

Root Chord = 2.5 ft.

Tip Chord = 1.67 ft.

Wing Flap:

Type: Plain

Root Chord = 1.58 ft.

Tip Chord = 1.25 ft.

Span = 9 ft.

Max. Deflection = 40°

Distance from root quarter chord to leading edge of horizontal tail = 170 in. = 14.15 ft.

Distance from root quarter chord to propeller plane = l_p = 7.64 ft.

Propeller diameter = 7 ft.

Thrust axis is parallel to fuselage reference line.

Elevator gearing, G = 1.05 rod/ft.

C_L = lift coefficient. Subscripts a, w, t denote airplane, wing, and tail, respectively.

C_m = moment coefficient. Subscript "ac" denotes moment coefficient about the aerodynamic center, subscript "FUS" denotes moment coefficient due to the fuselage, etc.

V = velocity, fps.

$\bar{V}$ = tail volume $= \frac{S_t}{S_w} \frac{l_t}{C} = .616$

T_c = thrust coefficient $= \frac{550 \eta_p SHP}{\rho V^3 D^2}$

where; η_p = propeller efficiency

SHP = shaft horsepower

ρ = air density, slugs/ft^3

V = airplane velocity, fps

D = propeller diameter, ft.

c.g. = center of gravity.

z = perpendicular distance from c.g. to thrust line, positive downward, ft.

α = angle of attack. Subscripts w, t, p, denote wing, tail, and propeller thrust axis respectively.

$\alpha_p = \alpha_w - 2^\circ$

X_{cg} = c.g. position in per cent of chord.

X_{ac} = a.c. position in per cent of chord.

$\frac{Xa}{C} = X_{cg} - Xac$ = distance from c.g. to a.c. in per cent of chord.

S_p = propeller disk area $= \frac{\pi D^2}{4} = 38.5$ ft^2

C_{np} = Propeller Normal force coefficient.

m = W/g, Mass.

μ = airplane relative density factor $= \frac{m}{\rho S_w c}$

τ = time constant = $\frac{m}{\rho S_w V}$

N_o = Control-fixed neutral point.

N_o' = Control-free neutral point.

N_m = Control-fixed maneuvering point.

N'_m = Control-free maneuvering point.

Flight Configurations Investigated

Clean Condition -- Power On:

V = 128 mph
SHP = 130
W = 2670 lbs.

Clean Condition -- Power Off:

V = 128 mph
SHP = 0
W = 2670

Landing Condition -- Power On:

V = 78 mph
SHP = 145
W = 2670

Landing Condition -- Power Off:

V = 78 mph
SHP = 0
W = 2670

The above conditions were investigated at center of gravity position, X_{cg} = .212 (Clean Condition).

Note: Lowering wheels for landing condition moves the center of gravity forward .5%.

Additional Data

In addition to the general airplane specifications the following data is presented:

a) $c_{mac} = -.0342$ for NACA 4415R.

When corrected for three dimensional flow: (Reference 1)

Clean Condition:

$$C_{mac} = Ec_{mac} = (1.08)\,(-.0342) = -.037$$

Landing Condition:

$$C_{mac} = Ec_{mac} + E'\Delta c_{mac} + C_{ml_b} =$$

$$-\,.037 + (.72)\,(-.19) + 0 \doteq -\,.174$$

b) $$\left(\frac{d\,C_L}{d\alpha}\right)_{Wing} = a_w = \frac{f\,a_o}{1 + \frac{57.3}{\pi A}\,a_o}$$ (Reference 1)

$$a_w = \frac{(.998)\,(.106)}{1 + \frac{57.3}{\pi}\,\frac{(.106)}{6.04}} = .08$$

c) $$a_t = \frac{A\,a_o}{AE_e + \frac{57.3}{\pi}\,a_o}$$ (Reference 2)

where; $AE_e = \frac{A}{A+2} + A = 4.7$

$$a_t = \frac{(4.02)\,(.108)}{4.7 + \frac{57.3}{\pi}\,(.108)} = .065$$

d) $$\alpha_{o\,LW} = \alpha_{o\,L_{root}} + J\epsilon$$ (Reference 1)

$$\epsilon = \alpha_{o\,L_{tip}} - \alpha_{o\,L_{root}} = -6 + 4 = -2^{\circ}$$

$$J = -\,.41$$

$$\alpha_{o\,LW} = -4 + (-.41)\,(-2) = -3.2^{\circ}$$

d) (Continued)

$$i_{root} = + 2^{\circ}$$

$$\alpha_{oL_{airplane}} = -5.2^{\circ} \qquad \text{(See Fig.I-1)}$$

e) Change in angle of zero lift with flap deflection:

$$\Delta \alpha_{oL} = -K \Delta C_l \qquad \text{(Reference 1)}$$

$$= -(6.7)\ (.88)$$

$$\Delta \alpha_{oL} = -5.9^{\circ} \qquad \text{(See Fig.I-1)}$$

f) Since propeller curves were lacking, Reference (4) was used to determine propeller characteristics. For the propeller efficiencies obtained, four per cent was arbitrarily subtracted for propeller surface roughness.

Clean Condition: $\eta_p = .78$

Landing Condition: $\eta_p = .68$

g) In Fig.I-2 downwash, ϵ, is plotted vs. wing angle of attack, α_w. The curves were drawn from calculations based on Reference (3).

Clean Condition: $1 - \frac{d\epsilon}{d\alpha} = .48$

Landing Condition: $1 - \frac{d\epsilon}{d\alpha} = .65$

h) Slipstream velocity ratio; $\left(\frac{V_s}{V}\right)^2$

Clean -- Power On: (Reference 1)

$$\left(\frac{V_s}{V}\right)^2 = 1 + \frac{8}{\pi} T_o = \eta_t$$

$$T_C = \frac{550\,\eta_p\ \text{SHP}}{\rho V^3\ D^2} = .078$$

(See "Flight Conditions Investigated.")

$$\left(\frac{V_s}{V}\right)^2 = 1 + \frac{8}{\pi}(.078) = 1.2$$

Since only approximately one-half the horizontal tail is immersed in the slipstream of the propeller it will be assumed that the effective velocity ratio is:

$$\left(\frac{V_s}{V}\right)^2 = 1 + \frac{8}{\pi}\,\frac{T_c}{2} = 1.1$$

An additional correction for fuselage boundary layer and interference effects must be applied as given in Reference 5. This correction is -.07 which gives a final value for velocity ratio:

$$\left(\frac{V_s}{V}\right)^2 = 1.1 - .07 = 1.03$$

Similarly;

Clean -- Power Off: $\left(\frac{V_s}{V}\right)^2 = .93$

Landing Condition: Power On: $\left(\frac{V_s}{V}\right)^2 = 1.3$

Power Off: $\left(\frac{V_s}{V}\right)^2 = .93$

i) $$\frac{(d\ C_m)}{(d\ CL)_{FUS}} = \frac{K_f\ W_f^{2}\ L_f}{S_w\ C\ a_w} = \frac{(.01)(16)(25)}{(184.2)(5.7)(.08)}$$ (Reference 1)

$$\frac{(d\ C_m)}{(d\ CL)_{FUS}} = .048$$

j) From Reference 6: $$C_{y_{\chi_o}} = \frac{d\ C_{np}}{d\ \alpha_p}$$

Clean Condition:

Power On: $$\frac{(d\ C_{np})}{(d\ \alpha_p)} = .1295, \text{ per degree}$$

Power Off: $$\frac{(d\ C_{np})}{(d\ \alpha_p)_{T_c\ =\ 0}} = .121, \text{ per degree}$$

Landing Condition:

Power On: $$\frac{(d\ C_{np})}{(d\ \alpha_p)} = .1015, \text{ per degree}$$

Power Off: $$\frac{(d\ C_{np})}{(d\ \alpha_p)_{T_c\ =\ 0}} = .0825, \text{ per degree}$$

k) $$\frac{d\ C_{np}}{d\ CL} = \frac{dC_{np}}{d\alpha_p}\ \frac{d\alpha_p}{d\ CL}\ ;\quad \frac{d\alpha_p}{d\ CL} = \frac{1 + \frac{d\epsilon}{d\alpha}}{a_w}$$ (Reference 1)

$$1 + \frac{d\epsilon}{d\alpha} = 1.15$$ (from upwash charts in Ref. 6)

$$\frac{d\alpha_p}{d\ CL} = \frac{1.15}{(.08)(57.3)} = .251$$

$$\frac{d\ C_{np}}{d\ CL} = (.251)\ \frac{d\ C_{np}}{d\alpha_p}$$

1) $$\frac{d\epsilon_p}{d\alpha} = A + B \frac{(d\,C_{np})}{(d\alpha\, p\,)_{T_c = 0}}$$ (Reference 6)

Clean -- Power On:

$$\frac{d\epsilon_p}{d\alpha} = .05 + (.26)(.121) = .082$$

$$\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right) = (.48 - .082) = .398$$

Clean -- Power Off:

$$\frac{d\epsilon_p}{d\alpha} = 0 + (.26)(.121) = .032$$

$$\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right) = (.48 - .032) = .448$$

Landing Condition -- Power On:

$$\frac{d\epsilon_p}{d\alpha} = .15 + (.26)(.0825) = .1714$$

$$\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right) = (.65 - .1714) = .479$$

Landing Condition -- Power Off:

$$\frac{d\epsilon_p}{d\alpha} = 0 + (.26)(.0825) = .0214$$

$$\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right) = (.65 - .0214) = .629$$

Sample Calculations for Stability Parameters

$$C_{mcg} = C_{L_W} \frac{Xa}{C} + C_{mac} + C_{mFUS} - C_{Lt} \bar{V} \left(\frac{v_s}{v}\right)^2 + T_c \frac{2D^2}{Sw} \frac{z}{C}$$

$$+ C_{np} \frac{l_p}{S_w} \frac{S_p}{C} = 0$$

$$C_{La} = C_{L_W} + C_{L_t} \frac{S_t}{S_w} \left(\frac{v_s}{v}\right)^2 .$$

Clean -- Power On:

$X_{cg} = .212\ C$

$X_{ac} = .242\ C$

$z = -.625$ ft.

$C_{L_a} = .365$

$T_c = .0781$

$$C_{L_a} = C_{LW} + C_{Lt} \left(\frac{43}{184.2}\right)(1.03) = .365$$

$$C_{mcg} = -.03\,(.365 - .24\,C_{Lt}) - .37 + (.048)(.365)$$

$$-C_{Lt}\,(.616)(1.03) + (.0781)\,\frac{(2)(7)^2}{(184.2)}\,\frac{(-.625)}{(5.7)}$$

$$+ \frac{(.1295)(.251)(.365)(7.64)(38.5)}{(184.2)(5.7)} = 0$$

$C_{Lt} = -.050$

$$\frac{dC_m}{dC_L} = \frac{Xa}{C} + \frac{dT_c}{dC_L}\frac{2D^2}{Sw}\frac{z}{C} + \frac{dC_{np}}{dC_L}\frac{l_p}{S_w}\frac{S_p}{C}$$

$$- \frac{a_t}{a_w}\bar{V}\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right)\left(\frac{v_s}{v}\right)^2 - C_{Lt}\ \bar{V}\ \frac{d\left(\frac{v_s}{v}\right)^2}{d\,C_L}$$

$$+ \frac{(d\ Cm)}{(d\ CL)_{FUS}}$$

$$\frac{d\left(\frac{v_s}{v}\right)^2}{d\ C_L} = \frac{8}{\pi}\frac{d\ T_c}{d\ C_L} = 1.74\ \eta_p\ C_{L_a}^{\frac{1}{2}}$$

$$= (1.74)(.78)(.365)^{\frac{1}{2}} = .82$$

$$\frac{dC_m}{dC_L} = -.03 + \frac{\pi}{8}\frac{(.82)(2)(7)^2(-.625)}{(184.2)(5.7)} + \frac{(.1295)(.251)(7.64)(38.5)}{(184.2)(5.7)}$$

$$- \frac{(.065)}{(.08)}(.616)(.398)(1.03) - (-.050)(.616)(.82) + .048$$

$$\frac{dC_m}{dC_L} = -.172$$

$$N_o = X_{cg} - \frac{dC_m}{dC_L} = .212 + .172 = .384$$

Clean -- Power Off

$$\frac{dC_m}{dC_L} = \frac{Xa}{C} + \frac{dC_{np}}{dC_L}\frac{l_p}{S_w}\frac{S_p}{C} - \frac{a_t}{a_w}\bar{V}\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon_p}{d\alpha}\right)\left(\frac{v_s}{v}\right)^2$$

$$+ \frac{(dC_m)}{(dC_L)_{FUS}}$$

$$\frac{dC_m}{dC_L} = -.03 + (.121)(.251)\frac{(7.64)(38.5)}{(184.2)(5.7)} - \frac{(.065)}{(.08)}(.616)(.448)(.93) + .048$$

$$\frac{dC_m}{dC_L} = -.181$$

Landing Condition -- Power On:

$X_{cg} = .207\ C$

$X_{ac} = .242\ C$

$z = -.75$ ft.

$C_{La} = .88$

$T_c = .29$

$$C_{mcg} = C_{L_W}\frac{X_a}{C} + C_{mac} + C_{mFUS} - C_{Lt}\,\overline{V}\left(\frac{v_s}{v}\right)^2 + T_c\frac{2\,D^2\,z}{S_W\,C}$$

$$+ C_{np}\frac{l_p}{S_W}\frac{S_p}{C} = 0$$

$$C_{La} = C_{L_W} + C_{Lt}\frac{(S_t)}{(S_W)}\left(\frac{v_s}{v}\right)^2 = .88$$

$$C_{L_W} = .88 - C_{Lt}\frac{(43)}{(184.2)}(1.3) = .88 - .303\,C_{Lt}$$

$$C_{mcg} = -.035\,(.88 - .303\,C_{Lt}) - .174 + (.048)(.88)$$

$$- C_{Lt}\,(.616)(1.3) + (.29)\frac{(2)(7)^2(-.75)}{(184.2)(5.7)} + \frac{(.1015)(.251)(.88)(7.64)(38}{(184.2)(5.7)}$$

$$= 0$$

$$C_{Lt} = -.224$$

$$\frac{dC_m}{dC_L} = \frac{Xa}{C} + \frac{dT_c}{dC_L}\;\frac{2D^2}{S_w}\;\frac{z}{C} + \frac{dC_{np}}{dC_L}\;\frac{l_p}{S_w}\;\frac{S_p}{C}$$

$$- \frac{a_t}{a}\;\bar{V}\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon\ p}{d\alpha}\right)\left(\frac{V_s}{V}\right)^2 - C_{Lt}\;\bar{V}\;\frac{d\left(\frac{V_s}{V}\right)^2}{d\ C_L} + \frac{(dC_m)}{(dC_L)_{FUS}}$$

$$= -.035 + (.488)\frac{(2)(7)^2(-.75)}{(184.2)(5.7)} + (.1015)\frac{(.251)(7.64)(38.5)}{(184.2)(5.7)}$$

$$- \frac{(.065)}{(.08)}(.616)(.479)(1.3) - (-.224)(.616)(1.11) + (.048)$$

$$\frac{dC_m}{dC_L} = -.178$$

Landing Condition -- Power Off:

$$\frac{dC_m}{dC_L} = \frac{Xa}{C} + \frac{dC_{np}}{dC_L}\;\frac{l_p}{S_w}\;\frac{S_p}{C} - \frac{a_t}{a_w}\;\bar{V}\left(1 - \frac{d\epsilon}{d\alpha} - \frac{d\epsilon\ p}{d\alpha}\right)\left(\frac{V_s}{V}\right)^2 + \frac{(dC_m)}{(dC_L)_{FUS}}$$

$$= -.035 + \frac{(.0825)(.251)(7.64)(38.5)}{(184.2)(5.7)} - \frac{(.065)}{(.08)}(.616)(.629)(.93) + .048$$

$$\frac{dC_m}{dC_L} = -.274$$

Elevator Hinge Moment Parameters:

(References 1,7 and 8)

$$c_{h_\alpha} = -.0048$$

$$c_{h_\delta} = -.011$$

$$C_{h_\alpha} = c_{h_\alpha} \frac{a}{a_o} = -.0048 \frac{(.08)}{(.106)} = -.0036$$

$$C_{h_\delta} = c_{h_\delta} + \tau (C_{h_\alpha} - c_{h_\alpha})$$

$$\tau = .54$$

$$C_{h_\delta} = -.011 + (.54) \left[-.0036 + .0048 \right] = -.0104$$

Elevator Power:

$$C_{m_\delta} = -a_t \bar{V} \eta_t \tau$$

$$= -(.065)(.616)(1.03)(.54)$$

$$C_{m_\delta} = -.0223 \qquad \text{(Clean -- Power On)}$$

$$N_o - N_o' = -\frac{C_{m_\delta} C_{h_\alpha}}{a_w C_{h_\delta}} \left(1 - \frac{d\epsilon}{d\alpha}\right) - \frac{HM_b C_{m_\delta}}{W/S \; C_{h_\delta} S_e \; c_e}$$

where: HM_b = elevator balance due to effective mass unbalance = 10 ft. lbs.

$$N_o - N_o' = \frac{-(-.0223)(-.0036)(.48)}{(.08)(-.0104)} - \frac{(10)(-.0223)}{(14.5)(-.0104)(15.04)(1.25)}$$

$$= .0463 - .0788 = -.0325$$

$$N_o' = .384 + .0325 = .417 \qquad \text{(Clean -- Power On)}$$

$$N_m = N_o - \frac{63 g\, l_t\, \rho\, C_{m\delta}}{2\; W/S}\left(1 + \frac{1}{n^2}\right) \qquad \text{(Steady turns)}$$

$$= N_o \; \frac{-(63)(32.2)(15.04)(.00196)(-.0223)}{(2)\;(14.5)}\left(1 + \frac{1}{(1.3)^2}\right)$$

$$= N_o + .073$$

$$N_m = .384 + .073 = .457 \quad \text{(Clean -- Power On)}$$

$$N_m' = N_o' + \frac{57.3\, g l_t \rho\, C_{m\delta}}{2\, W/S\;\; C_{h\delta}}\left(1 + \frac{1}{n^2}\right)\left(C_{h\alpha} - \frac{1.1\, C_{h\delta}}{\tau}\right)$$

$$N_m' = N_o' + \frac{(57.3)(32.2)(15.04)(.00196)(-.0223)}{(2)(14.5)(-.0104)}(1.59)\left(-.0036 - \frac{(1.1)(-.0104)}{.54}\right)$$

$$= N_o' + .113$$

$$N'_m = .417 + .113 = .530 \quad \text{(Clean -- Power On)}$$

Clean -- Power On:

$$C_{md\alpha} = -a_t\, \bar{V} \eta_t\, \frac{1}{\mu}\, \frac{l_t}{c}\, \frac{d\epsilon}{d\alpha}$$

$$\mu = \frac{m}{\rho S c} = \frac{2670}{(32.2)(.00196)(184.2)(5.7)} = 40.4$$

$$C_{md\alpha} = \frac{-(.065)(.616)(1.03)(15.04)(.52)}{(40.4)(5.7)} = -.00140 \text{ per degree} = -.08 \text{ per radian}$$

$$C_{md\theta} = -1.1\;\; a_t\;\; \bar{V} \eta_t\;\; \frac{l_t}{c}\;\; \frac{1}{\mu}$$

$$= \frac{-(1.1)(.065)(.616)(1.03)(15.04)}{(5.7)(40.4)}$$

$$C_{md\theta} = -.00296 \text{ per degree}$$

$$= -.17 \text{ per radian}$$

$$C_{m\alpha} = C_{L\alpha}\,(X_{cg} - N_o) \qquad X_{cg} = .3\, C$$

$$= (.08)(-.08)(57.3) = -.37 \text{ per radian}$$

Airplane lift curve slope:

$$C_{L_\alpha} = \frac{a_w + a_t \frac{S_t}{S_w} \eta_t \left(1 - \frac{d\epsilon}{d\alpha}\right)}{1 - \frac{(dC_m)}{(dC_L)_{FIX}} \quad \frac{c}{l_t}}$$

Clean -- Power Off at Mid -c.g.:

$$\frac{(dC_m)}{(dC_L)_{FIX}} = -.09$$

$$C_{L_\alpha} = \frac{.08 + (.065)\frac{(43)}{(184.2)}(.93)(.48)}{1 - \frac{(5.7)}{(15.04)} (-.09)} = .084 \text{ per degree}$$

Table I - 1

Tabulated Results

	Clean Condition		Landing Condition	
Parameter	Power On	Power Off	Power On	Power Off
C_{L_α}		.084		
C_{m_δ}	-.022	-.020	-.028	-.020
N_o	.38	.39	.39	.48
N_o'	.42	.42	.41	.50
N_m	.46	.46		
N'_m	.53	.52		
C_{m_α} (based on N_o)	-.37			
C_{m_α} (based on N_m)	-.75			
$C_{md\theta}$	-.17	-.15		
$C_{md\alpha}$	-.08	-.073		

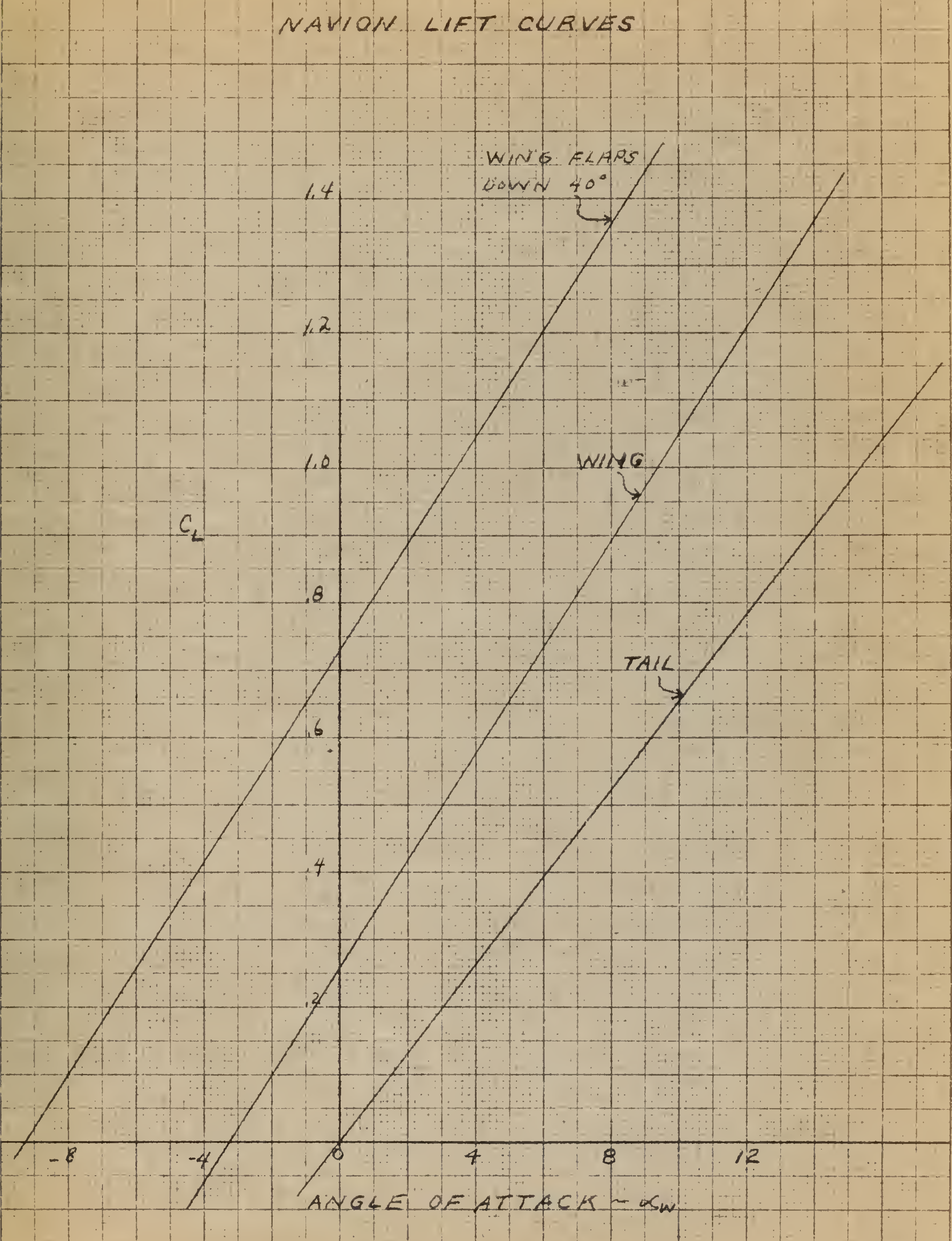
NAVION LIFT CURVES
WING FLAPS DOWN 40°
WING
TAIL
C_L
1.4
1.2
1.0
.8
.6
.4
.2
-8
-4
0
4
8
12
ANGLE OF ATTACK ~ α_W

FIG. I-1

DOWNWASH ANGLE
VS.
ANGLE OF ATTACK

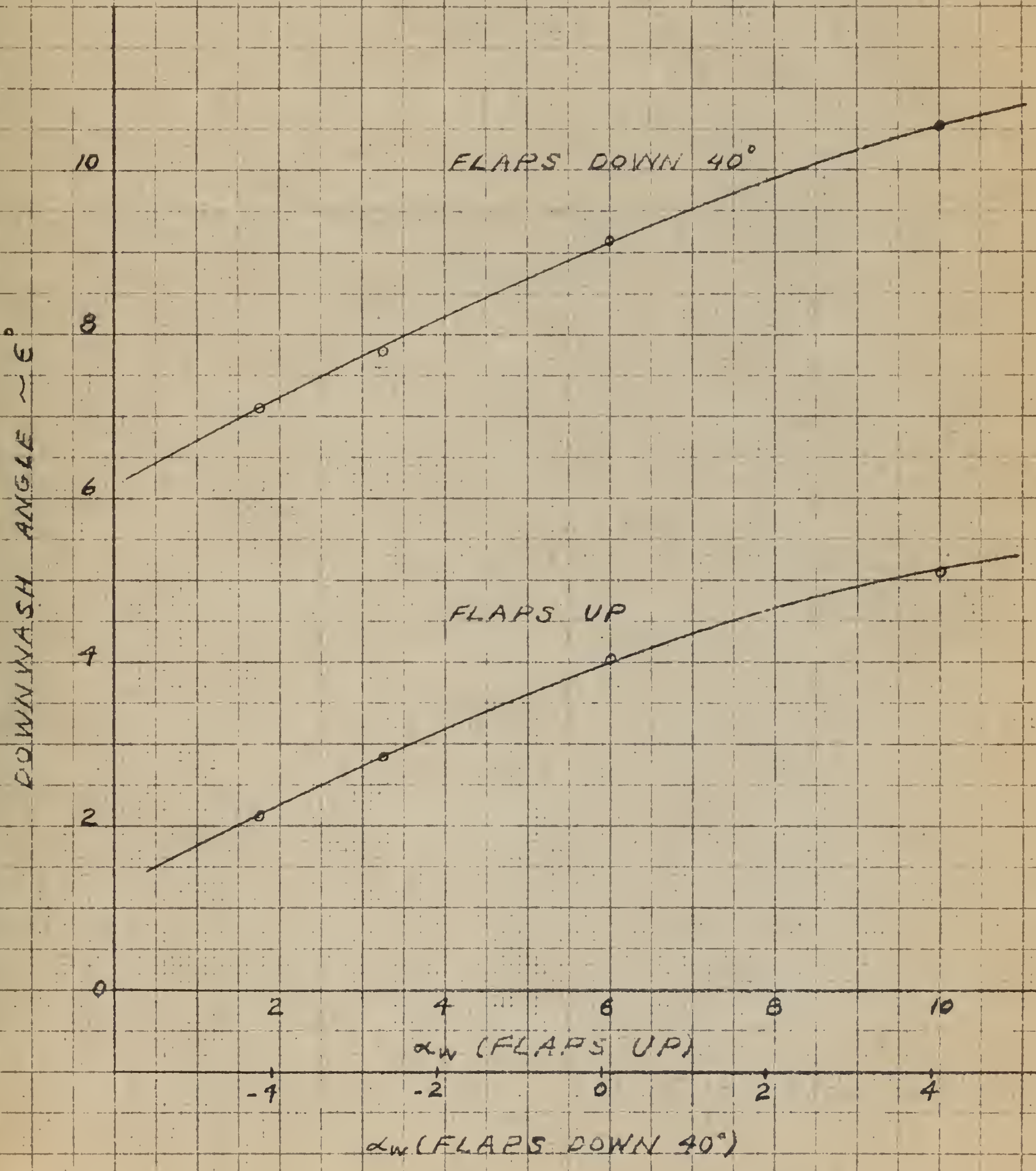

FIG. I-2

PART II

DETERMINATION OF AERODYNAMIC PARAMETERS BY STEADY STATE FLIGHT TESTING METHODS

PART II

DETERMINATION OF AERODYNAMIC PARAMETERS BY STEADY STATE FLIGHT TESTING METHODS

All of the steady state flight testing was done in Ryan Navion, N5113K. The steady state tests involved 20 hours of flying time and covered a period of three months, February 13 to May 13, 1953.

Due to the fact that additional instrumentation for the dynamic tests was being installed and also because additional weight was required to move the center of gravity to the rear positions, the maximum gross weight of the airplane increased from 2670 to 2875 lbs. during the period over which the tests were conducted. This increase and also the weight of the fuel consumed during each test were considered in the determination of the lift coefficients in the process of data reduction.

Center of gravity changes were made by moving and securing lead weights to the aircraft's structure to obtain a range from 21.2 to 37.8 percent of the mean aerodynamic chord. The lowering of the landing gear and flaps moved the center of gravity forward 0.5 percent.

The steady state flight project may be considered in three areas, Flight Test Procedure, Flight Test Data Analysis and Instrumentation.

FLIGHT TEST PROCEDURE

Static Stability Tests:

Flight tests were conducted with two external configurations and two power conditions for each configuration. The power on, clean configuration was trimmed at 128 miles per hour with a constant power setting which produced an average of 130 horsepower. The power settings were adjusted at the beginning of each test to compensate as much as possible for atmospheric conditions. The power off, clean configuration was trimmed at 107.5 miles per hour using approximately 10 inches of manifold pressure, a power setting which simulated a feathered propeller.

The landing configuration was flown with the landing gear and flaps full down and the cowl flaps closed. The trim speed for both power on and power off landing conditions was 78 miles per hour.

The aircraft was trimmed at the mid altitude of the altitude range over which the tests were flown. Normally the range extended 1000 ft. above and below the mid altitude which was 6500 ft. The trim tab was not adjusted after the initial trim for each test was made.

The aircraft was steadied in each attitude and corresponding airspeed at which data were obtained. Stick force, elevator angle, angle of attack and airspeed readings were taken at each point and altitude, temperature and fuel level readings were taken periodically during the tests.

Maneuvering Stability Tests:

Only the clean configuration was tested for maneuvering stability. The power on and off conditions corresponded to those used for the static stability tests. The same trim procedure and mid altitude were also used. The range of altitude over which the tests were conducted extended about 1500 ft. above and below the mid altitude. this increase in altitude range was due to the greater rate of descent and longer time required to steady the aircraft in maneuvering flight.

Normal acceleration was experienced by placing the aircraft in a left bank while maintaining a constant airspeed of 128 miles per hour for the power on condition and 107.5 miles per hour for the power off condition. An increase in bank yielded a corresponding increase in normal acceleration. Elevator angle and stick force readings were taken by the observer while the pilot read the acceleration. Periodic temperature, altitude and fuel level readings were made.

In general, the flight test procedures followed corresponded to those perscribed in Part II of the Naval Flight Test Manual, Ref. 9.

FLIGHT TEST DATA ANALYSIS

The flight test data were reduced as indicated in Ref. 9. An example of static stability data reduction is as follows:

$W_{max.}$ = 2875 lbs.

Wing Area = S = 184.2 sq. ft.

C.G. position = 37.8% M.A.C.

Condition - Clean

Weight of fuel = 6 lbs./gal.

Fuel consumed to reach test altitude = 14 gal.

Fuel consumed during test = 3 gal.

Average aircraft weight during test = 2875 - 15.5 x 6 = 2782 lbs.

Dynamic pressure = $q = 0.00256\, V^2_{cal.(mph.)}$

Lift coefficient = $C_L = \frac{W}{qS} = \frac{2782}{184.2q} = \frac{15.1}{q}$

at 124 mph.;

$q = .00256\,(124)^2 = 39.4$

$C_L = \frac{15.1}{39.4} = .383$

The elevator angle, angle of attack and stick force reduction merely consisted of applying the calibration curves to the readings obtained during the flight test.

The reduction of the maneuvering stability was similar to that described above with the following exception:

$C_L = \frac{nW}{qS}$; where n = normal acceleration.

Figs. II-2 through II-5 are plots of the elevator angle versus the airplane lift coefficient for the static stability tests. The elevator power, C_{m_δ} , was determined for the various configurations and power conditions from these curves by the relationship:

$$C_{m_\delta} = - C_L \frac{\Delta x_{ac}}{\Delta \delta_e};$$

where Δx_{ac} is the change in center of gravity position divided by the mean aerodynamic chord to render it dimensionless; and $\Delta \delta_e$ is the change in elevator angle encountered at constant lift coefficient over the center of gravity range used. C_L is the lift coefficient at which the change in elevator angle is measured.

An example of the calculation involved is as follows:

Condition - Power on, landing.

$C_L = .88$

at c.g. position = 37.3% m.a.c.; $\delta_e = 12.65$ degrees,

at c.g. position = 20.7% m.a.c.; $\delta_e = 4.55$ degrees.

$\Delta \delta_{e_o} = 3.0$ degrees

$\Delta x_{ac} = .166$

$\Delta \delta_e = 12.65 - 4.55 - 3 = 5.10$ degrees

$$C_{m_\delta} = -.88 \frac{(.166)}{(5.10)} = -.0286$$

The difference in values of elevator angle for zero lift, δ_{e_o}, is believed due to the different settings of the trim tab required to trim at the same speed for the four center of gravity positions

at which the tests were conducted. It is believed that the effect of the tab is merely a translation of the δ_e versus C_L curve to a higher δ_{e_o} for the more rearward center of gravity positions. On this basis, the values of C_{m_δ} were determined considering the curves to be translated to such an extent that a common value of δ_{e_o} exists for the four curves.

Figs. II-6 through II-8 are static stability plots of stick force divided by dynamic pressure versus airplane lift coefficient for the various center of gravity positions. Some rather unexpected curvatures for this type of plot were encountered for which no complete explanation can be given. This was, perhaps, partially due to the relatively low stick forces which may have been erroneously indicated near the trim condition due to a certain amount of friction in the control system. It is not believed that the airplane possesses non-linearities to the extent indicated. However, the overall results compare favorably with the theoretical calculations, indicating that these data are representative.

Figs. II-9 and II-10 represent the variation of elevator angle with normal acceleration and lift coefficient, respectively, for maneuvering flight. Both plots are provided since there was a variation of weight for the various center of gravity configurations and the lift coefficient can no longer be expressed as the normal acceleration multiplied by a constant over the entire center of gravity range. The curves are linear since it was not feasible

to obtain a high enough normal acceleration to get noticeable non-linear elevator action.

Figs. II-11 and II-12 indicate the variation of stick force with normal acceleration in maneuvering flight. A small amount of curvature at the higher accelerations and more rearward centers of gravity indicates the normal trend.

The stick fixed neutral points for both clean and landing configurations are indicated on Fig. II-13. The neutral points are invarient with lift coefficient since the elevator angle versus lift coefficient curves are linear.

Figs. II-14 and II-15 show the location of the stick free neutral points. More stability is indicated as lift coefficient increases in the power off condition while the reverse is true for power on flight.

Figs. II-16 and II-17 indicate the stick fixed and stick free maneuvering points. The trends appear normal. Both the stick free neutral point and the stick free maneuver point are located behind their stick fixed counterparts due to the statically unbalanced elevator installed on this aircraft.

The airplane lift coefficient versus angle of attack is plotted on Fig. II-18 for one center of gravity location. The variation of lift curve slope with center of gravity position is shown on Fig. II-17. Since the variation should normally be linear, the points were faired

in this manner. The non-linear location of the points is believed due to calibration difficulties and the relatively large effect of scattered points on the determination of the lift curve slope. The general trend, however, appears proper.

$C_{m_{d\theta}}$, $C_{m_{d\alpha}}$ and $C_{m_{\alpha}}$ were calculated using flight test data in equations developed from theory. The evaluation of these derivatives was done as follows:

The elevator angle is related to the elevator angle at zero lift, the stick fixed stability margin, $C_{m_{\delta}}$ and the damping in pitch in accelerated flight as indicated below. This relationship is set forth in Ref. 1. The stability margin for $C_L V^2$ = constant, was assumed to be approximately equal to the stability margin for V = constant.

$$\delta_e = \delta_{e_0} - \frac{C_L}{C_{m_\delta}} \left(\frac{dC_m}{dC_L}\right)_{\substack{\text{FIXED} \\ (V=\text{Const.})}} - 63 \frac{C_{m_{\dot{\theta}}}}{C_{m_\delta}} \dot{\theta}$$

$$\delta_e = \delta_{e_0} - \frac{nW}{SqC_{m_\delta}} \left(X_{CG} - N_0\right)_{V=\text{Const.}} - 63 \frac{C_{m_{d\theta}}}{C_{m_\delta}} \Upsilon \frac{g}{V} \left(n - \frac{1}{n}\right)$$

$$\delta_e = \delta_{e_0} - \frac{W}{SqC_{m_\delta}} \left[\left(X_{CG} - N_0\right) n + 63 \frac{C_{m_{d\theta}}}{2} \left(n - \frac{1}{n}\right)\right]$$

$$\frac{d\delta_e}{dn} = -\frac{W}{SqC_{m_\delta}} \left[\left(X_{CG} - N_0\right) + 63 \frac{C_{m_{d\theta}}}{2} \left(1 + \frac{1}{n^2}\right)\right]$$

For clean condition, power on;

$N_o = 37\%$ m.a.c.

$\frac{d\delta_e}{dn} = -2.25$

$(x_{cg} - N_o)_{C_L V^2 = \text{const.}} = -.046$

$W_{avg.} = 2682$ lbs.

$n = 1.3$

$C_L = \frac{nW}{qS} = .347n$

$q = 42.0$

$C_{m_\delta} = -.0226$

$$-2.25 = \frac{-.347}{-.0226}\left[(-.046 + 50.25\, C_{m_{d\theta}})\right]$$

$$C_{m_{d\theta}} = \frac{-.1466 + .046}{50.25} = -.002 \text{ / degree or } -.1146\text{/radian}$$

$$C_{m_{d\alpha}} = C_{m_{d\theta}} \frac{d\epsilon}{d\alpha}$$

$\frac{d\epsilon}{d\alpha} = .52$ from theory

Therefore, for clean condition, power on;

$$C_{m_{d\alpha}} = -.1146\,(.52) \doteq -.0595$$

$$C_{m_\alpha} = \frac{dC_m}{dC_L}\frac{dC_L}{d\alpha} = a(x_{cg} - N_o)$$

For power on, clean condition, center of gravity location = 32.4%;

$$C_{m_\alpha} = 57.3\,(.106)(-.061) = -.370$$

The aerodynamic parameters obtained from steady state flight testing for the configurations and power conditions investigated are listed in Table II-1. Where parameters vary over a range of lift coefficients or normal accelerations, the value of that variable used in the calculation is indicated.

Table II-1

	Clean Condition		Landing Condition	
Parameter	Power On	Power Off	Power On	Power Off
C_{L_α}	.101	.090	.104	.1025
C_{m_δ}	-.0226	-.0219	-.0286	-.0244
N_o	.370	.385	.383	.413
N_o'	.435	.438	.433	.495
N_m	.448	.474	-----	-----
N_m'	.592	.506	-----	-----
$C_{m_{d\theta}}$	-.1146	-.130	-----	-----
$C_{m_{d\alpha}}$	-.0595	-.0675	-----	-----
C_{m_α}	-.370	-.253	-.407	-.570

Average values of C_L were taken at a center of gravity position at 26 percent of the mean aerodynamic chord.

$C_{L(clean)} = .356$ $n = 1.3$

$C_{L(landing)} = .88$

$C_{m_{d\theta}}$, $C_{m_{d\alpha}}$ and C_{m_α} calculations correspond to a center of gravity position = 32.4 percent of the mean aerodynamic chord.

INSTRUMENTATION

Power Supply:

Storage batteries, connected to furnish 24 volts, provided the power supply. The autosyn instruments were excited by 40 volts, alternating current, which was obtained through the use of a 400 cycle, 115 volt, inverter followed by a high frequency variac.

Stick Force Measurement:

Four Baldwin SR-4, type A-7, strain gages were installed on a wheel built for stick force measurement and were connected to form a bridge circuit as indicated in Fig. II-1. The voltage to the strain gages was maintained at 15 volts throughout the tests. The bridge balance provided for the centering of the microammeter for zero stick force. The sensitivity was adjusted to and maintained at 20 lbs. per 100 microamperes. The calibration of the installed system was very nearly linear.

Angle of Attack Measurement:

The angle of attack indicator was installed on a boom mounted on the left wing tip and extended four feet ahead of the leading edge of the wing. The vane was connected to the Pioneer AY-101D autosyn transmitter by a 12 to 1 gear ratio. A small plate was attached to the shaft of the vane within the head of boom.

Damping was provided by the movement of the plate through the oil, with which the head was filled. A Pioneer AY-5 autosyn repeater was mounted in the cockpit to give angle of attack indications. A relatively high excitation voltage, 40 volts, was used on both the angle of attack and elevator angle systems. This voltage setting appeared to reduce the repeater lag and needle oscillations almost entirely.

The angle of attack calibration was linear with the exception of the correction for position error. The position error was determined by measurement of the actual angle of attack in level flight with a propeller protractor.

Elevator Angle Measurement:

A 17 to 1 ratio of elevator movement to autosyn repeater movement was obtained by the use of a horn mounted on the elevator spar center flange and connected to the autosyn transmitter by waxed nylon line. The line was wrapped around a half inch pulley installed on the autosyn transmitter shaft and was secured to the trailing edge of the vertical fin by means of a light wire spring. Several turns of the chord were required around the pulley to prevent slippage. A Pioneer AY-5 autosyn repeater gave elevator angle readings in the cockpit.

The elevator position system calibration was linear. The calibration changed slightly over the period during which the flight

tests were made but remained constant during any one flight. This was determined by calibration before and after flights. Calibrations made after each flight were used to reduce the data obtained.

Acceleration Measurement:

A mechanical accelerometer was built and used to obtain maneuvering flight data. The accelerometer consisted of a weighted metal tube suspended on a spring and mounted inside a glass tube. The position of the upper edge of the metal tube in relation to the calibration mounted on the glass tube provided for a direct reading of acceleration. Reasonably accurate readings to .01 g could be made. The accelerometer was suspended from the upper cockpit structure to facilitate reading by the pilot.

General:

A sensitive airspeed indicator was used. Other instruments used were of normal configuration. In general, the instrumentation was relatively simple in nature and trouble free.

CONCLUSIONS AND RECOMMENDATIONS

Steady state flight testing is the accepted means of determining the airplane stability parameters in all cases where sufficient flight time to obtain the required data is available. Steady state testing methods are obviously not applicable to missiles or extreemly high speed aircraft which possess only limited endurance.

Steady state flight testing requires a fairly high quality of pilot technique and does require several more hours of flight time than comparable dynamic methods but more information may be obtained using the steady state methods.

The instrumentation required is relatively inexpensive and trouble free and the data reduction required is straight forward and relatively error free providing good data were obtained.

It is recommended that:

1. Careful and frequent calibrations be made of the instrumentation.

2. The angle of attack measuring instruments be calibrated more carefully.

3. Maneuvering flights be conducted in the same configurations and power conditions using several different airspeeds to get constant acceleration curves from which, perhaps, a better determination of the damping derivatives may be made.

References

1) Perkins, C.D., and Hage, R.E., Airplane Performance, Stability and Control, (John Wiley and Sons, 1950).

2) NACA TN 1581; "Relations and Charts for Low Speed Stability Derivatives of Swept and Unswept Wings," by Toll, T.A., and Queijo, M.J.

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4) NACA Report 640; "The Aerodynamic Characteristics of Full Scale Propellers Having 2, 3, and 4 Blades of Clark Y and R.A.F. 6 Airfoil Sections," by Hartman, E.P., and Bierman, D.

5) NACA WR L-761; Effect of Propeller Operation on the Pitching Moments of Single-Engine Monoplanes," by Goett, H.S. and Pass, H.R.

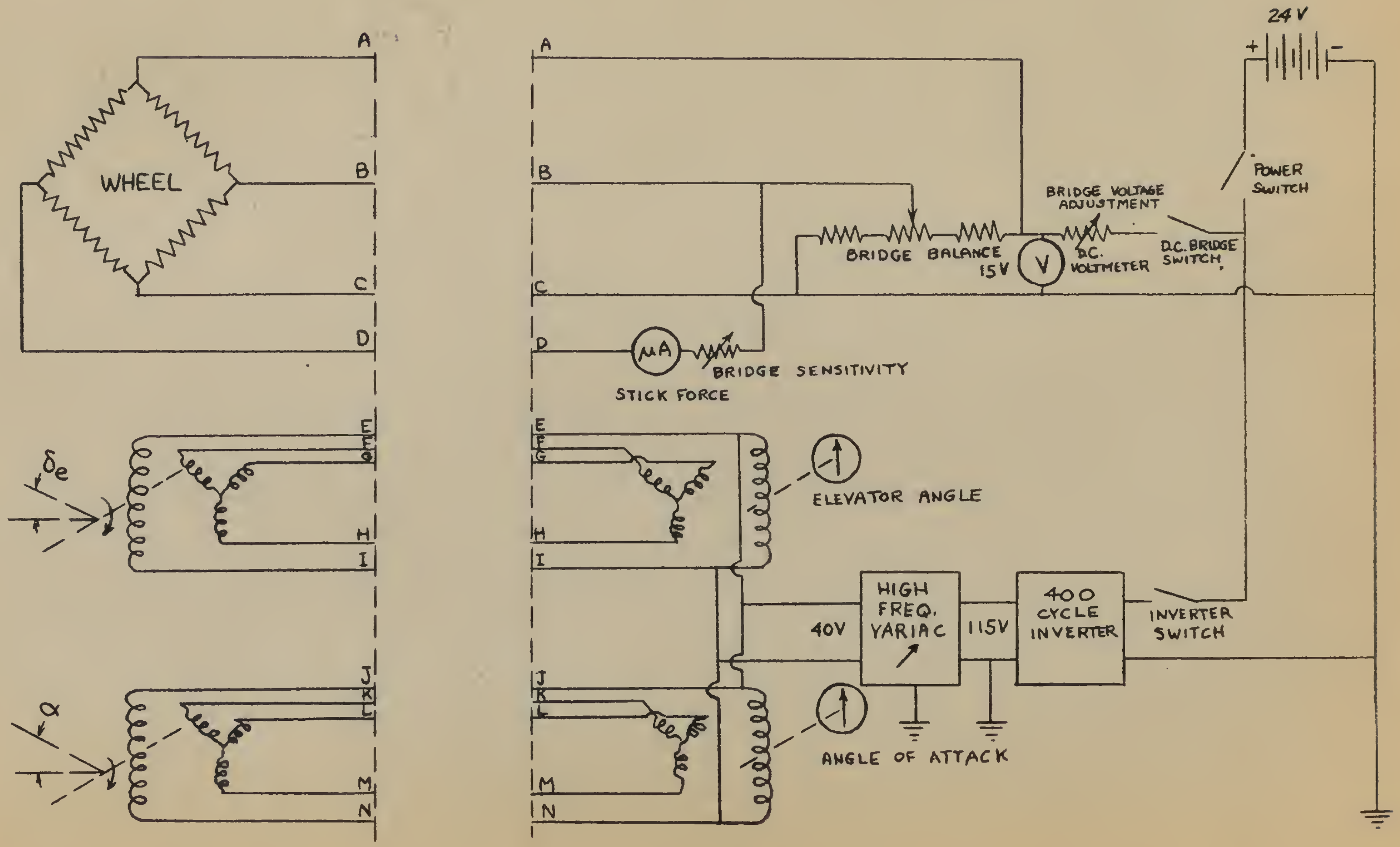
6) NACA WR L-25; "Notes on the Propeller and Slipstream in Relation to Stability," by Ribner, H.S.

7) NACA WR L-663; "Wind Tunnel Data on the Aerodynamic Characteristics of Airplane Control Surfaces," by Sears, R. I.

8) Dommasch, D.O., Sherby, S.S., and Connolly, T.F., Airplane Aerodynamics, (Pitman Publishing Corp. 1951).

9) Manual of Flight Test Methods and Procedures, Part II; U. S. Naval Air Test Center, Flight Test Division, Patuxent River, Maryland.

STEADY STATE FLIGHT INSTRUMENTATION NAVION N5113K Fig. II-1.



ELEVATOR ANGLE VERSUS AIRPLANE LIFT COEFFICIENT

CLEAN CONDITION
POWER ON
NAVION N5113K
ALTITUDE – 6500 FT

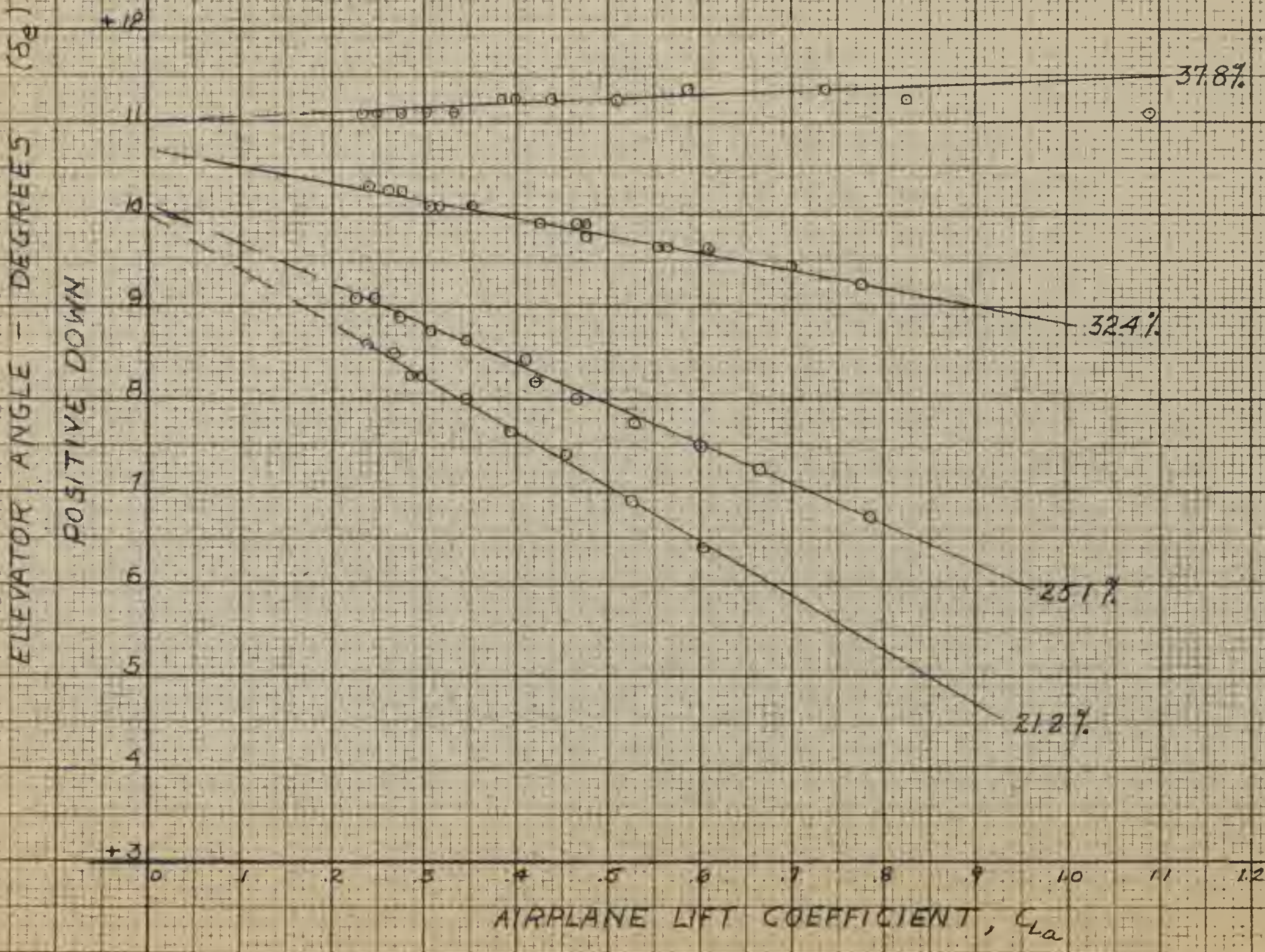


Fig. II-2

ELEVATOR ANGLE VERSUS AIRPLANE LIFT COEFFICIENT

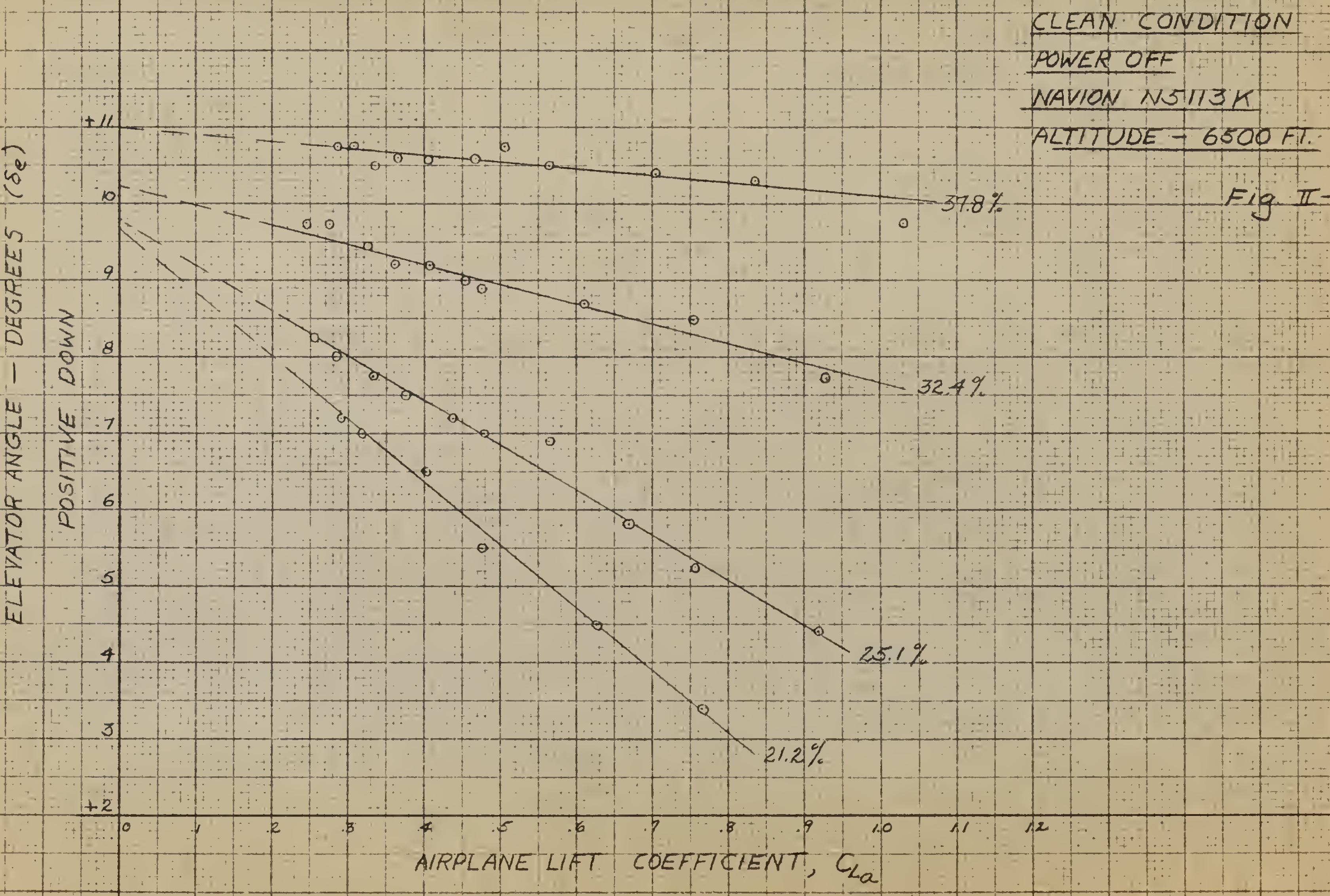
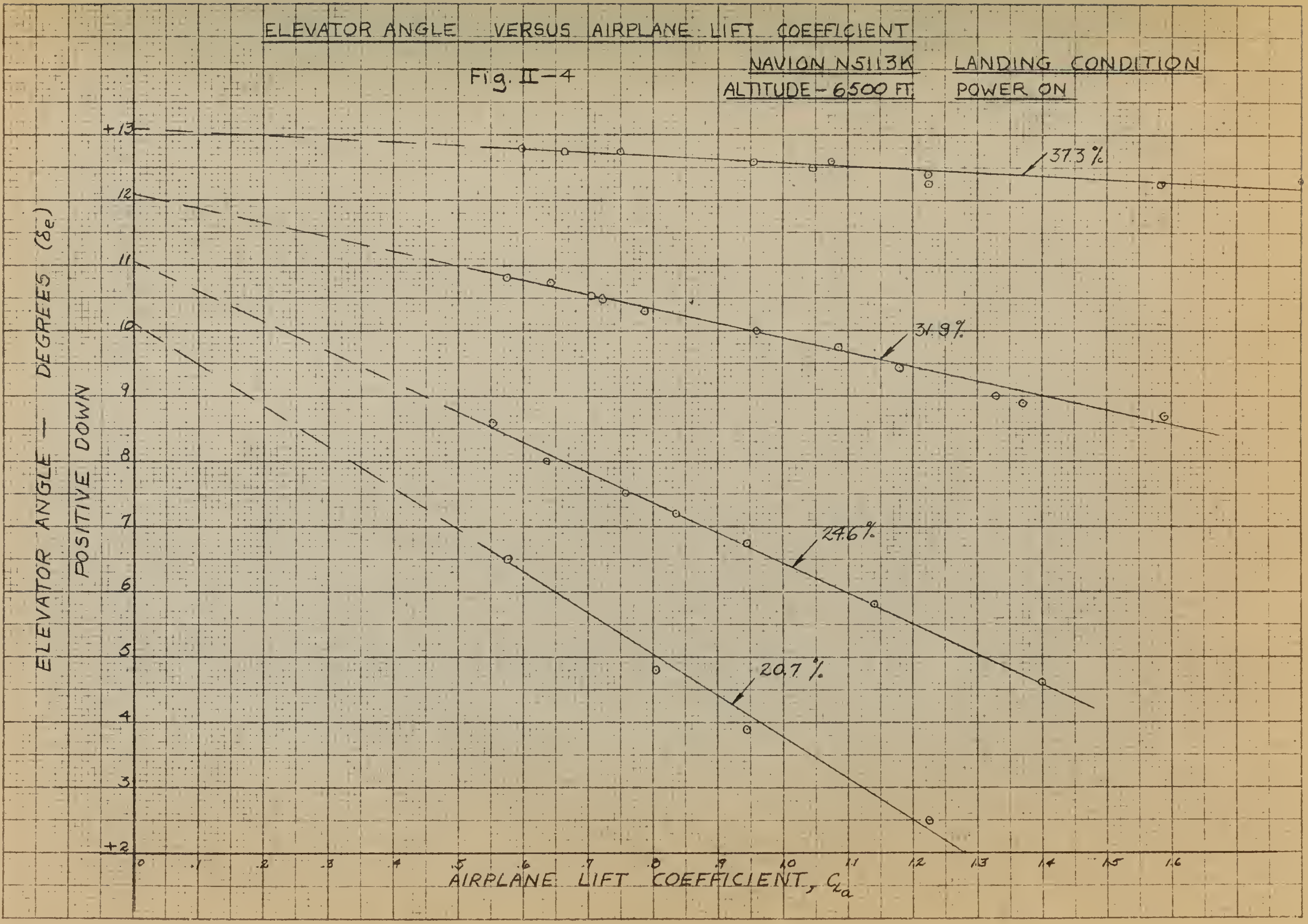


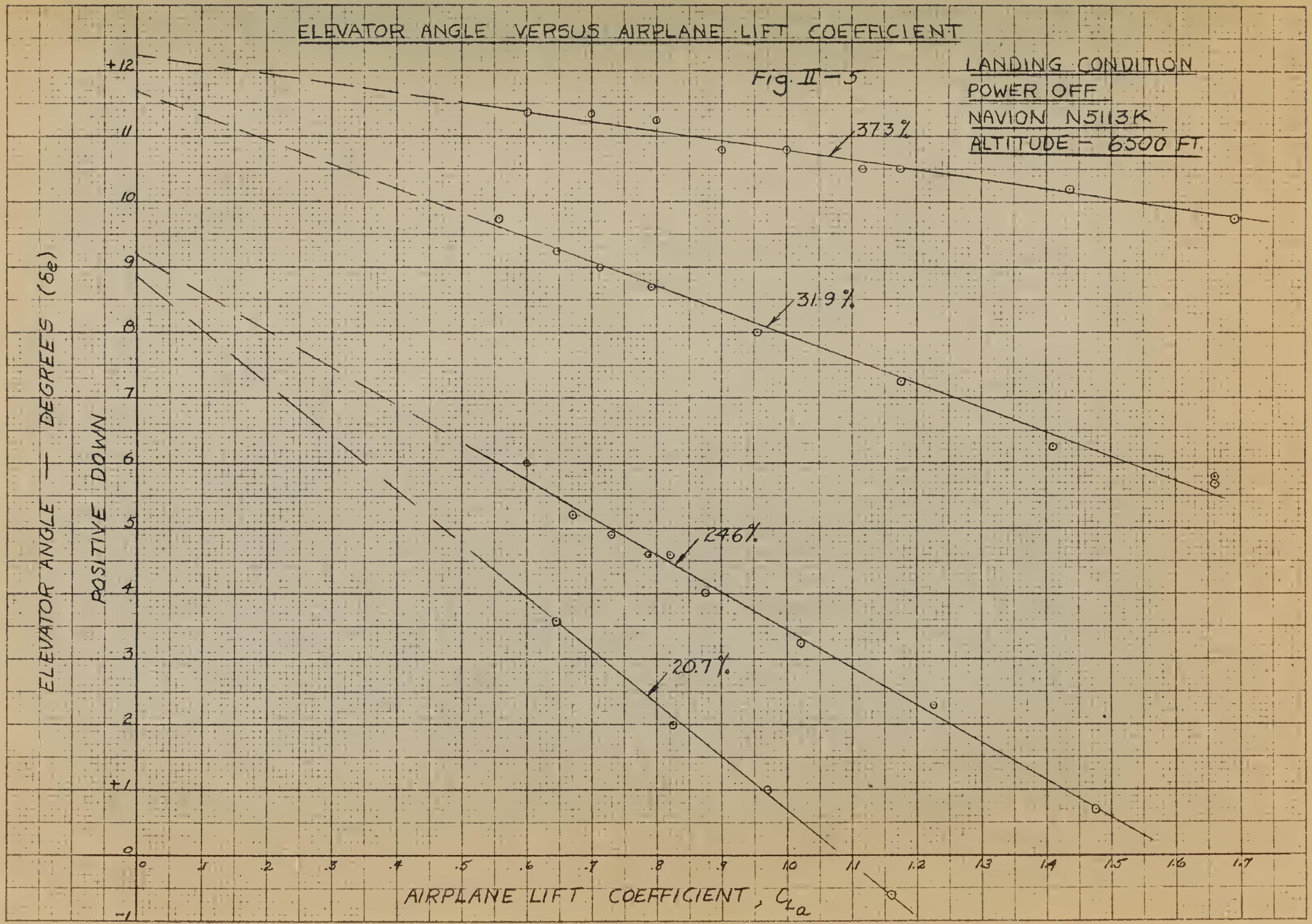
Fig. II—3



ELEVATOR ANGLE VERSUS AIRPLANE LIFT COEFFICIENT

Fig. II-4

NAVION N5113K LANDING CONDITION
ALTITUDE - 6500 FT. POWER ON



ELEVATOR ANGLE VERSUS AIRPLANE LIFT COEFFICIENT

Fig. II-5

LANDING CONDITION
POWER OFF
NAVION N5113K
ALTITUDE – 6500 FT.

STICK FORCE/DYNAMIC PRESSURE VERSUS AIRPLANE LIFT COEFFICIENT

POWER ON – CLEAN CONDITION
NAVION N5113K
ALTITUDE – 6500 FT.

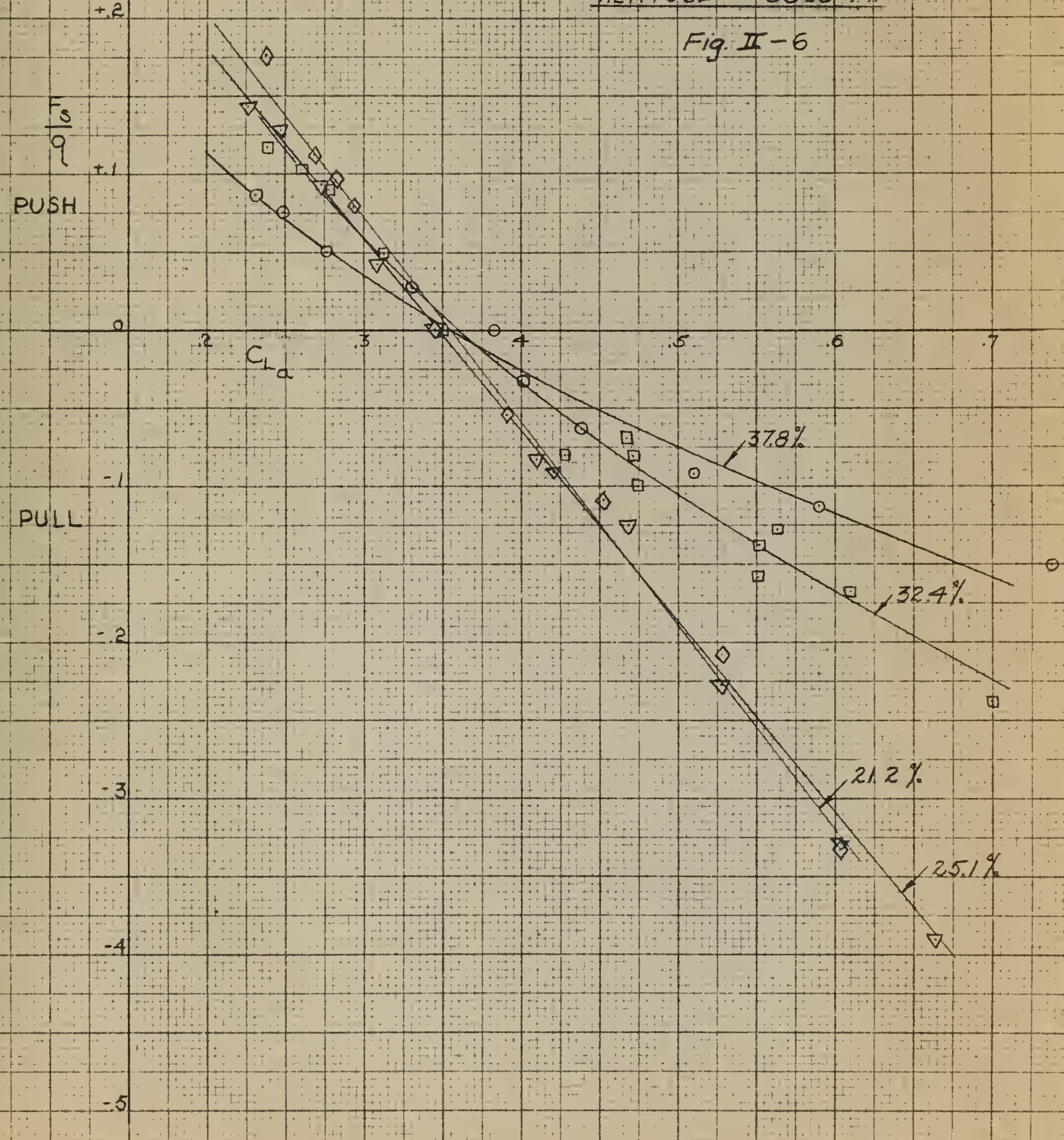
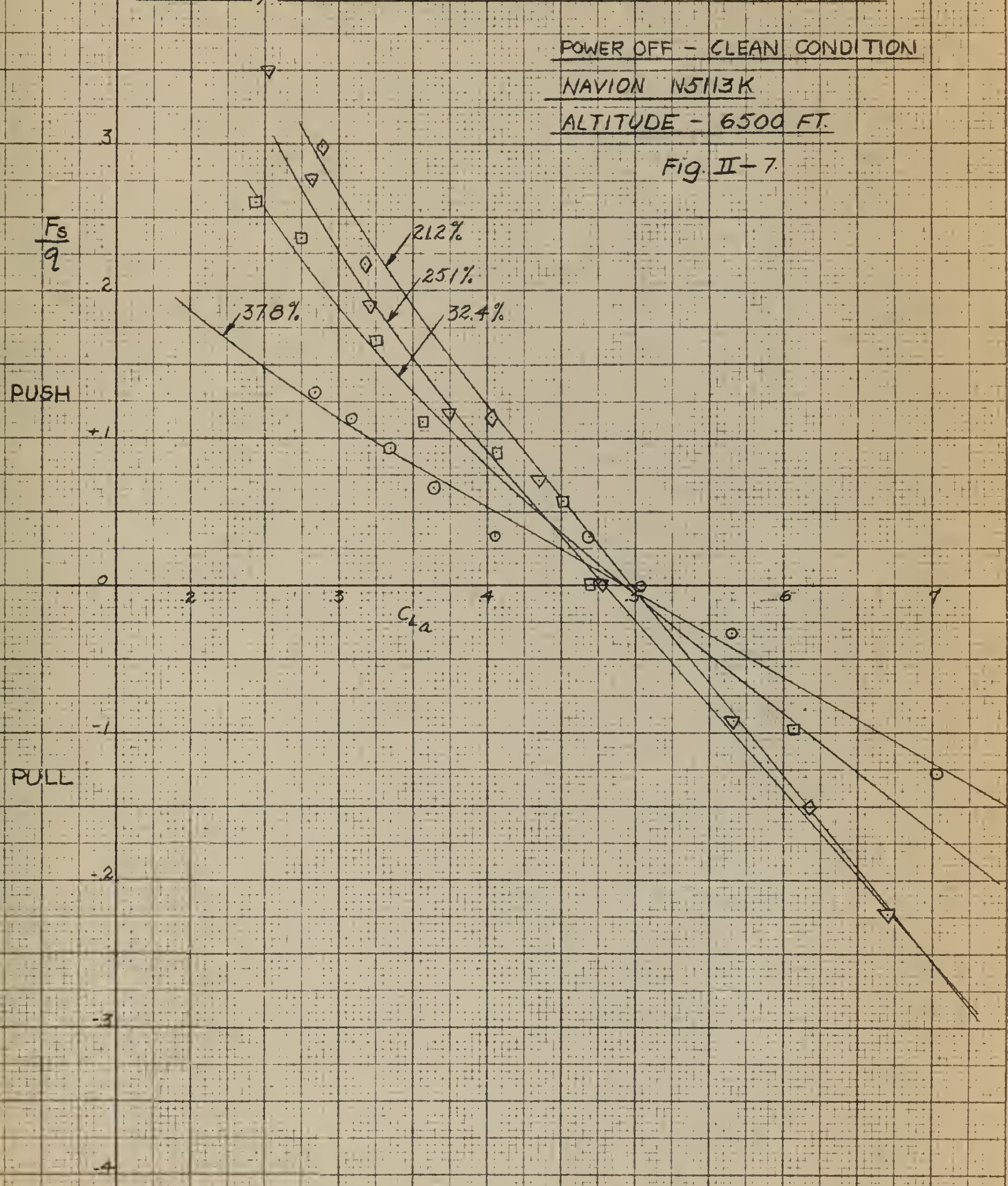


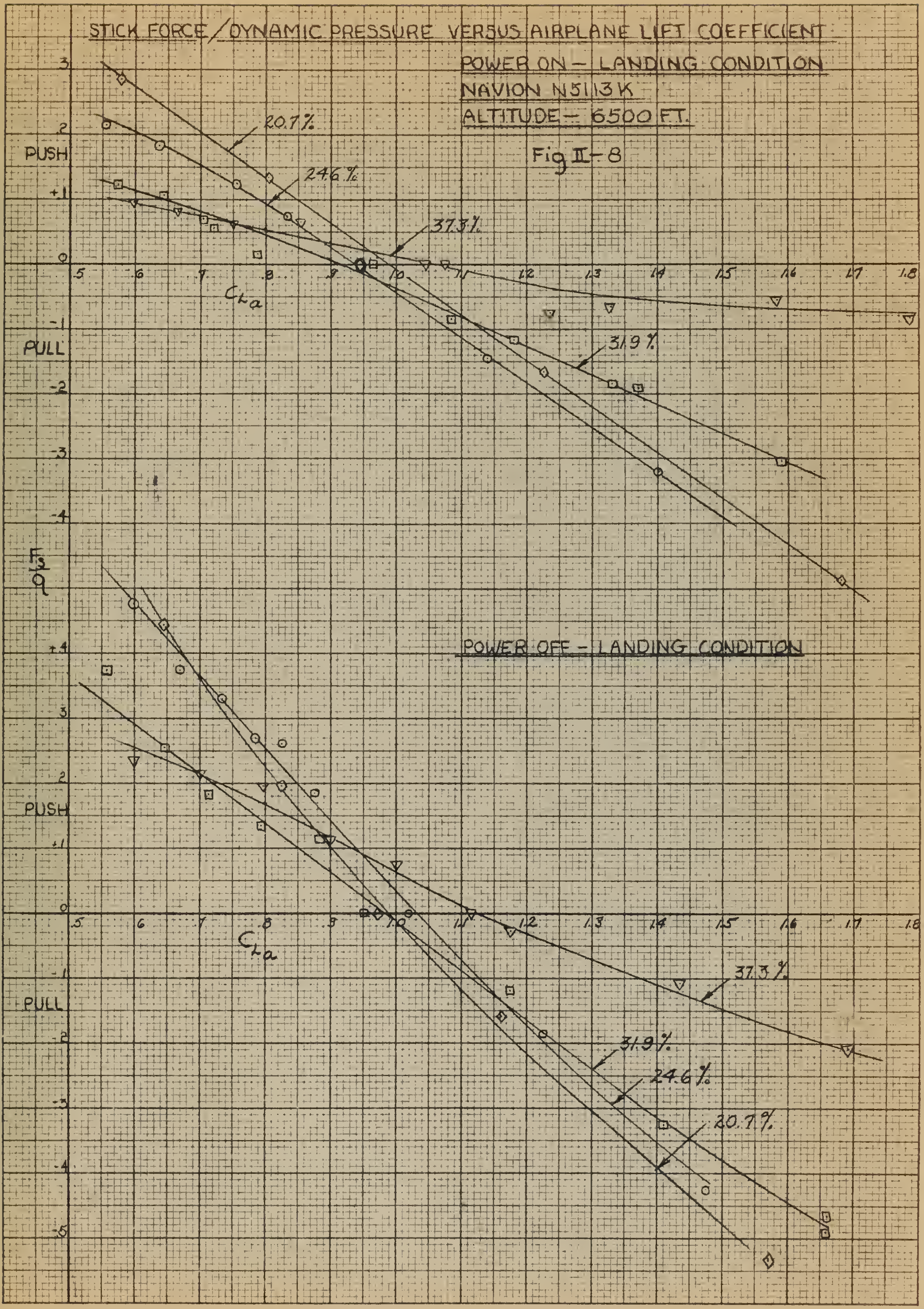
Fig. II – 6

STICK FORCE/DYNAMIC PRESSURE VERSUS AIRPLANE LIFT COEFFICIENT



POWER OFF - CLEAN CONDITION
NAVION N5113K
ALTITUDE - 6500 FT.

Fig. II-7



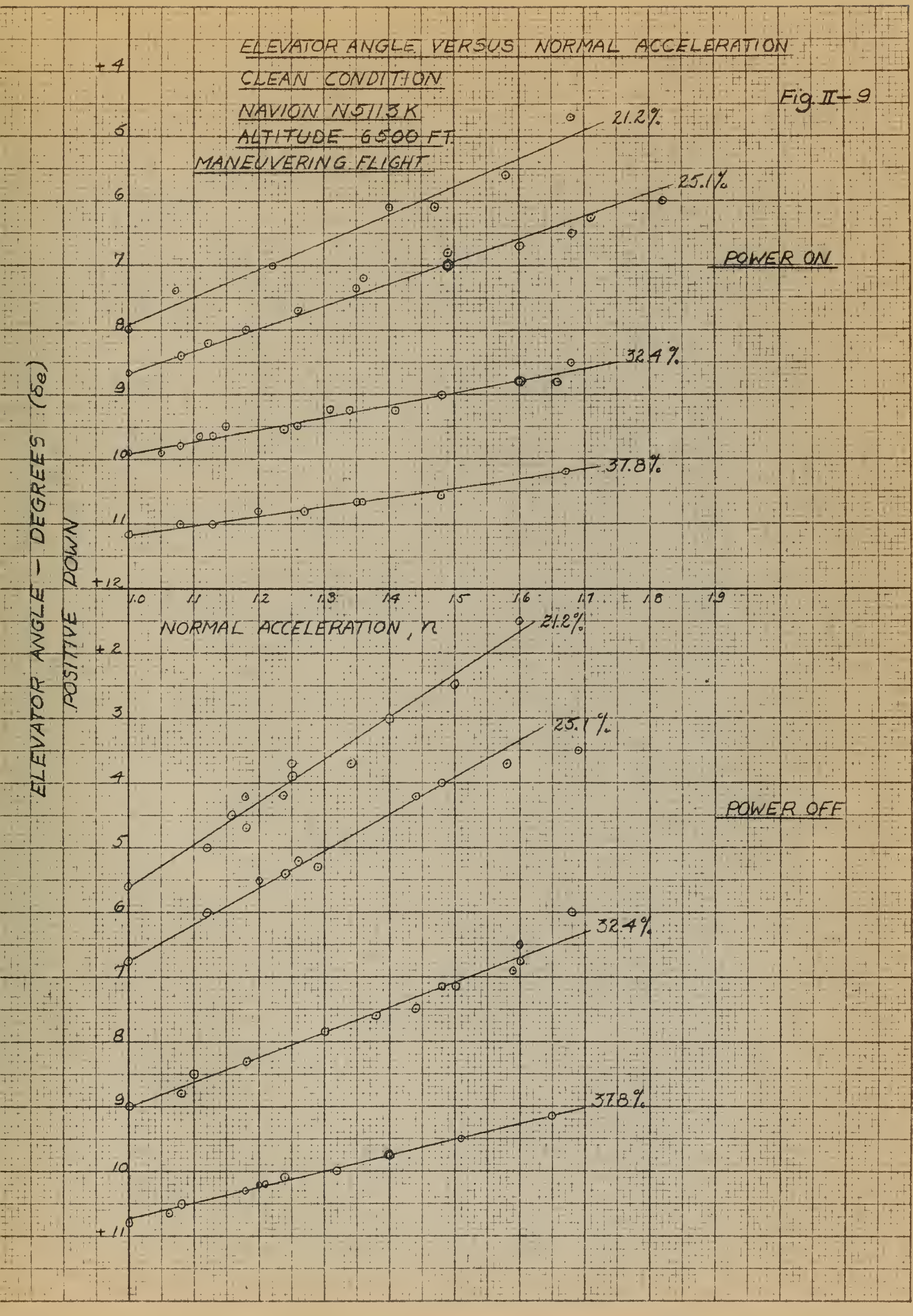
STICK FORCE/DYNAMIC PRESSURE VERSUS AIRPLANE LIFT COEFFICIENT

POWER ON – LANDING CONDITION

NAVION N5113K

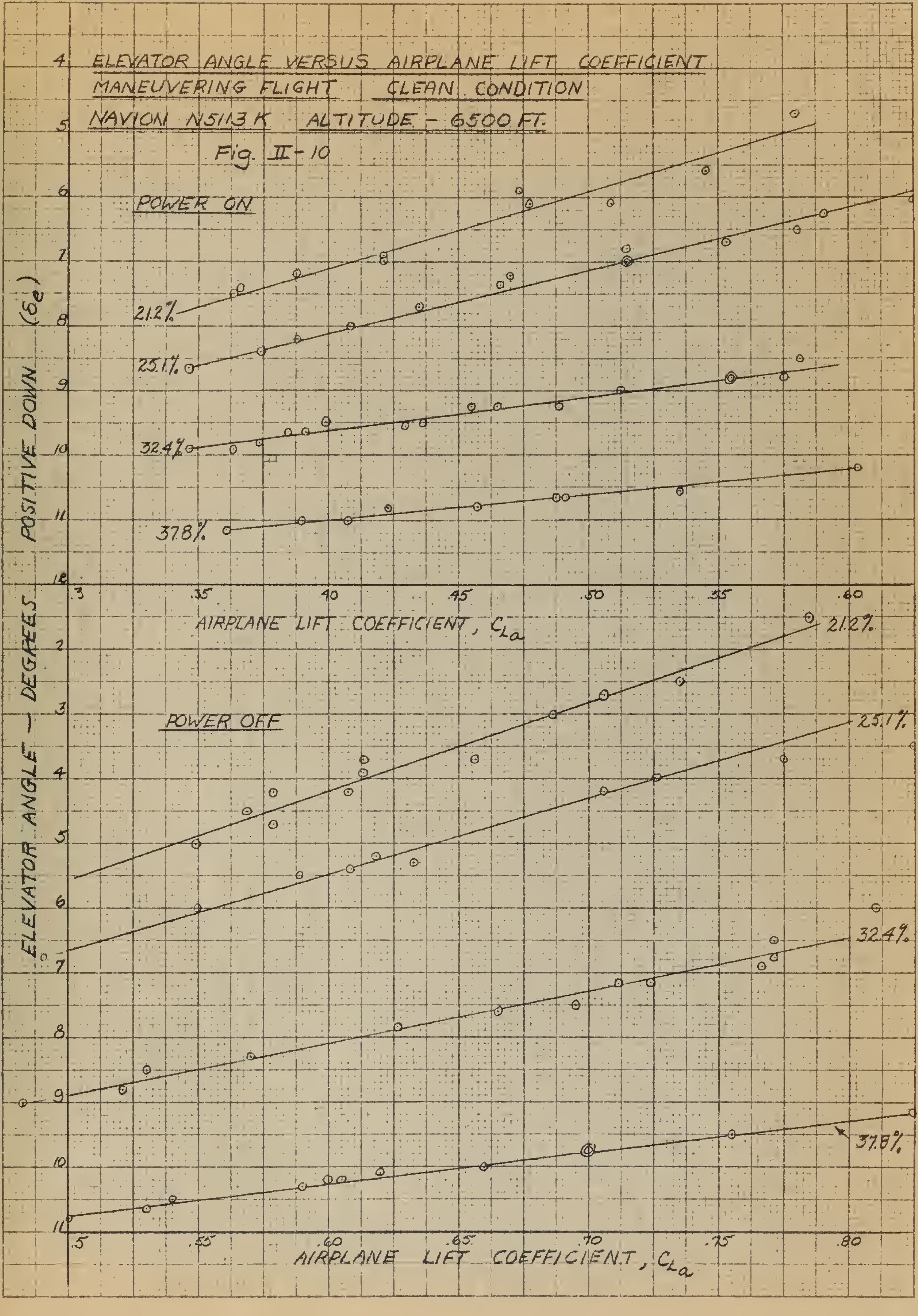
ALTITUDE – 6500 FT.

Fig II-8



ELEVATOR ANGLE VERSUS NORMAL ACCELERATION
CLEAN CONDITION
NAVION N5113K
ALTITUDE 6500 FT.
MANEUVERING FLIGHT

Fig. II-9



ELEVATOR ANGLE VERSUS AIRPLANE LIFT COEFFICIENT
MANEUVERING FLIGHT CLEAN CONDITION
NAVION N5113K ALTITUDE - 6500 FT.

Fig. II-10

STICK FORCE VERSUS NORMAL ACCELERATION

CLEAN CONDITION
POWER OFF
NAVION N5113K
ALTITUDE - 6500 FT.

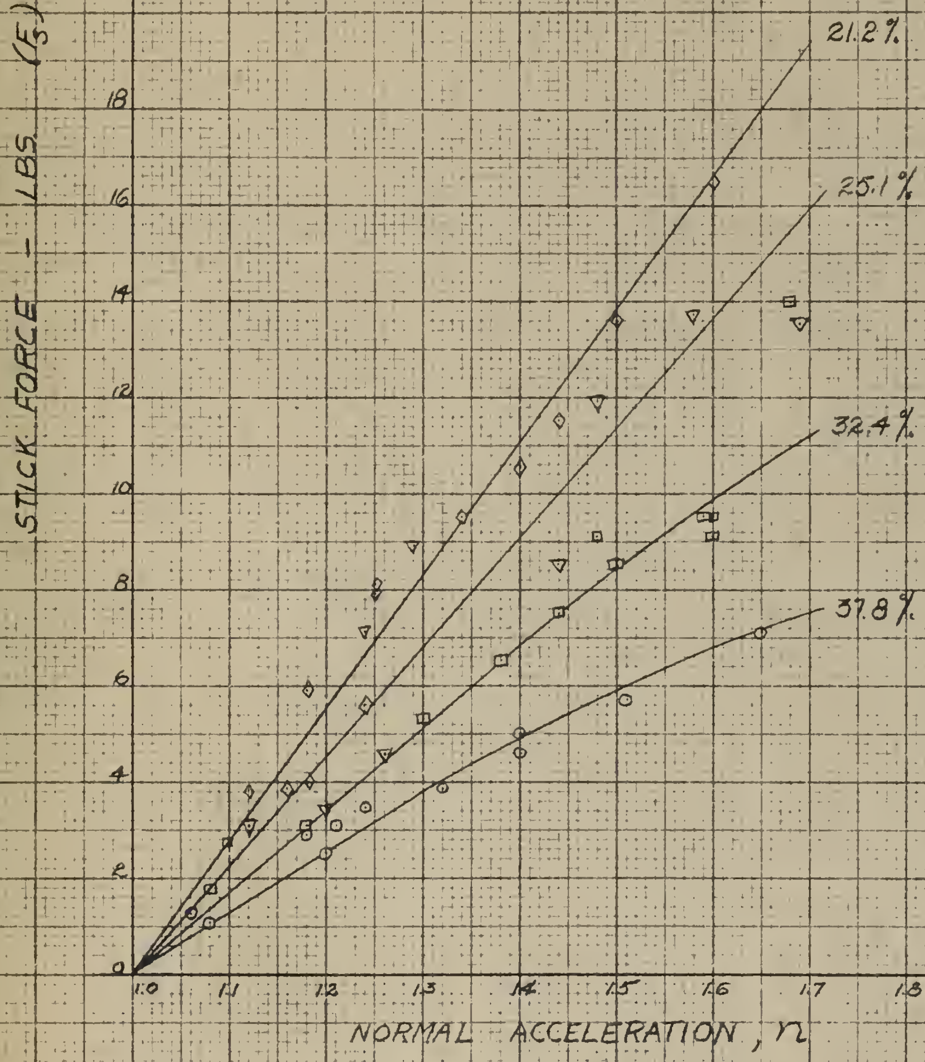
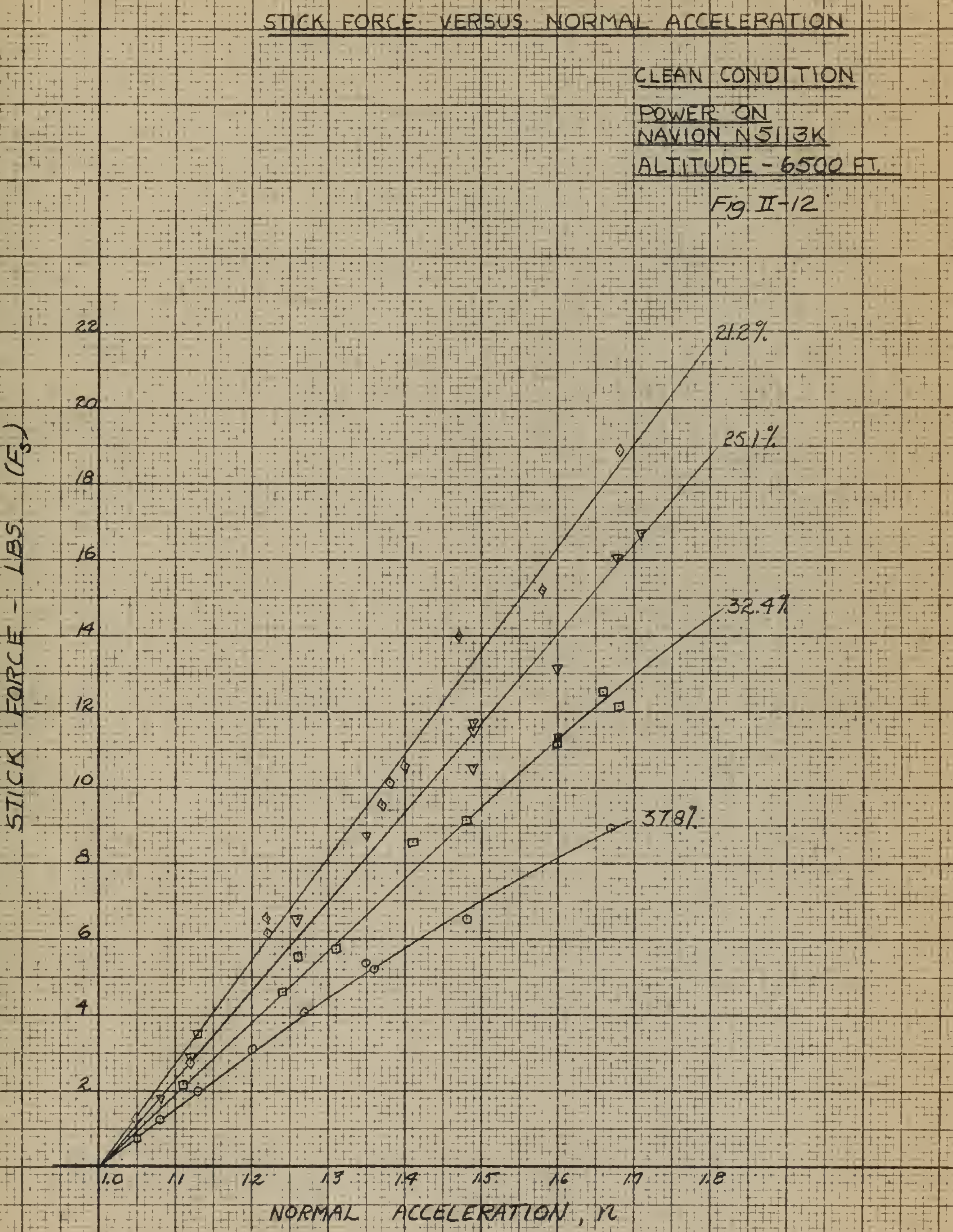


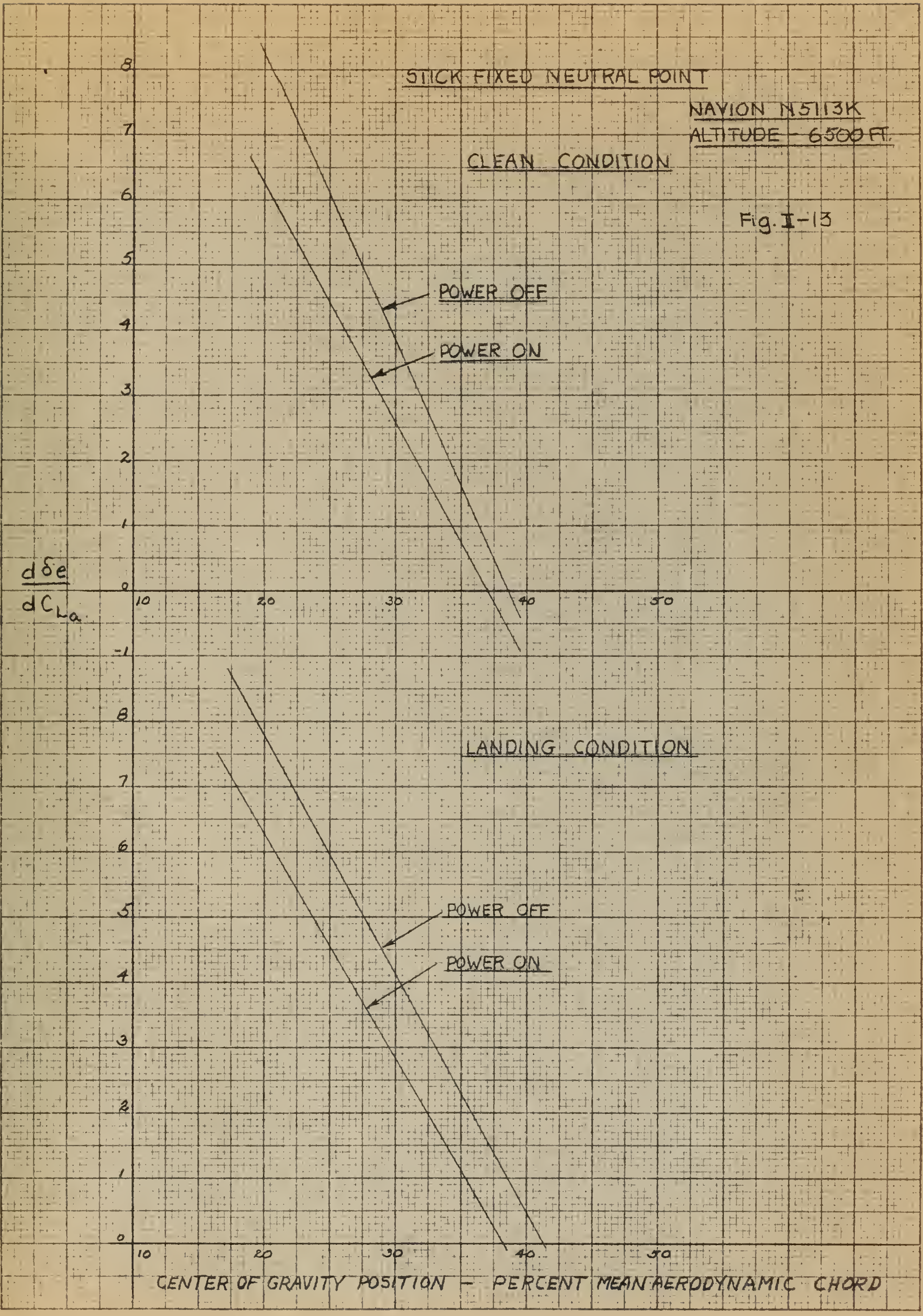
Fig II-11



STICK FORCE VERSUS NORMAL ACCELERATION

CLEAN CONDITION
POWER ON
NAVION N5113K
ALTITUDE - 6500 FT.

Fig. II-12



STICK FIXED NEUTRAL POINT

NAVION N5113K

ALTITUDE - 6500 FT.

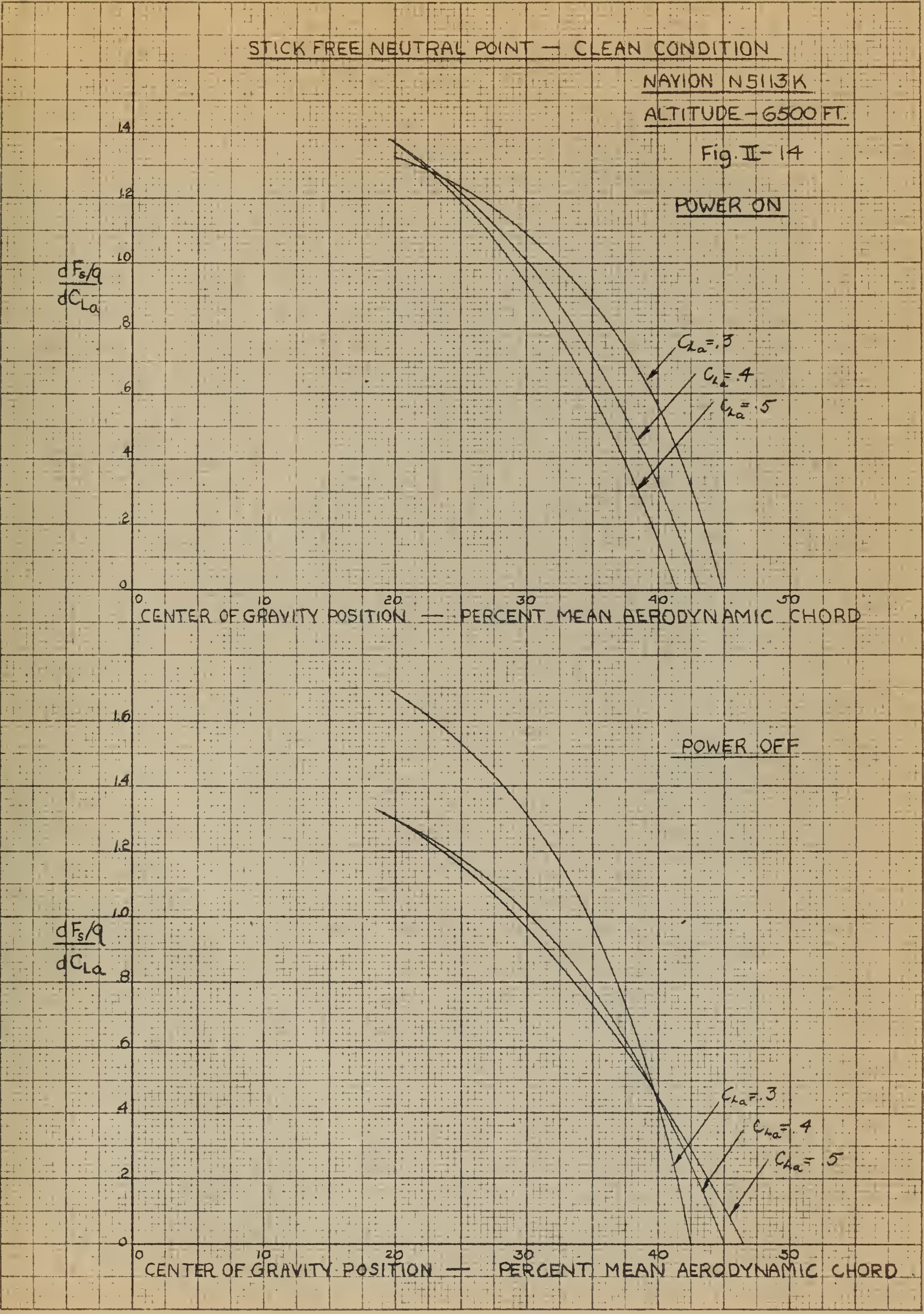
Fig. II-13

STICK FREE NEUTRAL POINT — CLEAN CONDITION

NAVION N5113K

ALTITUDE — 6500 FT.

Fig. II-14

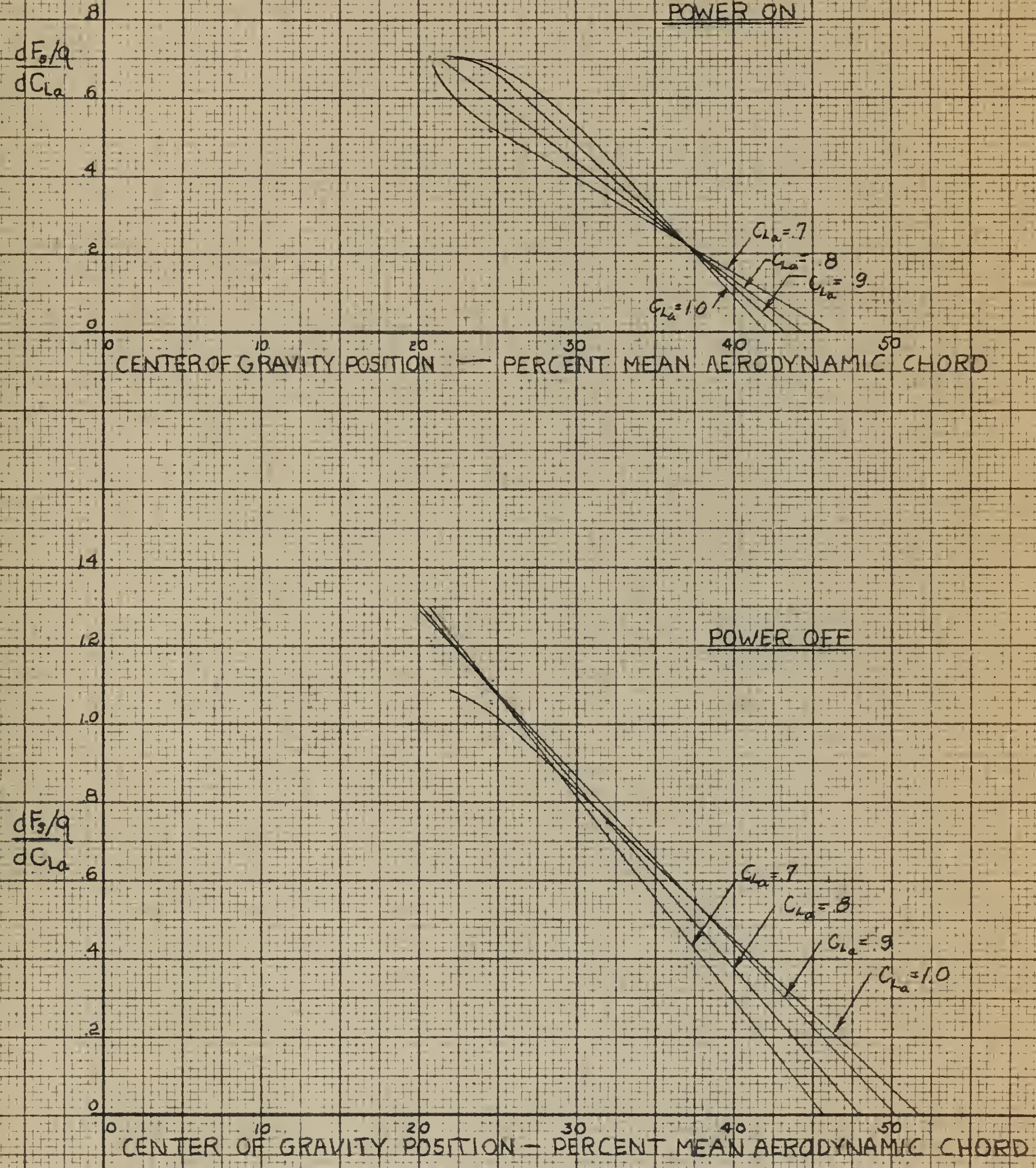


STICK FREE NEUTRAL POINT-LANDING CONDITION

NAVION N5115K

ALTITUDE - 6500 FT.

Fig. II-15



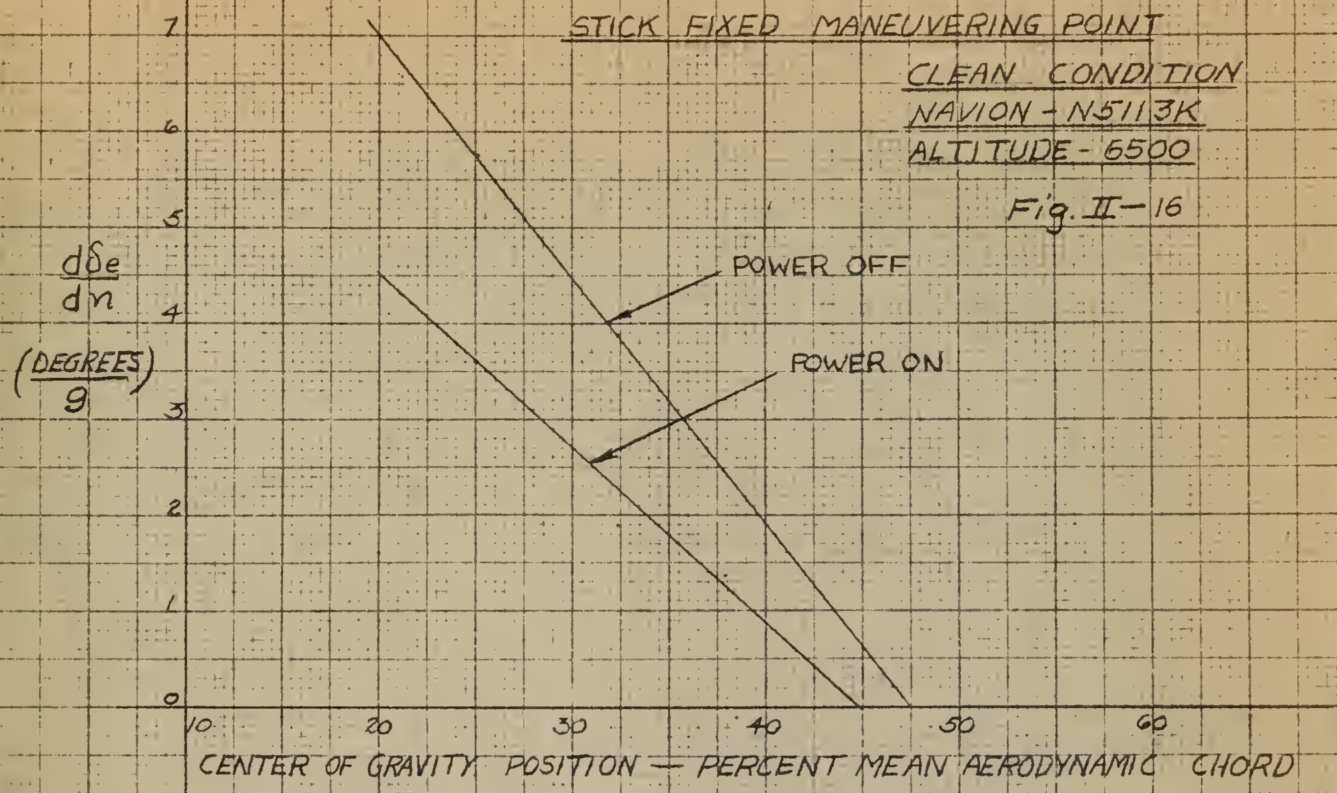
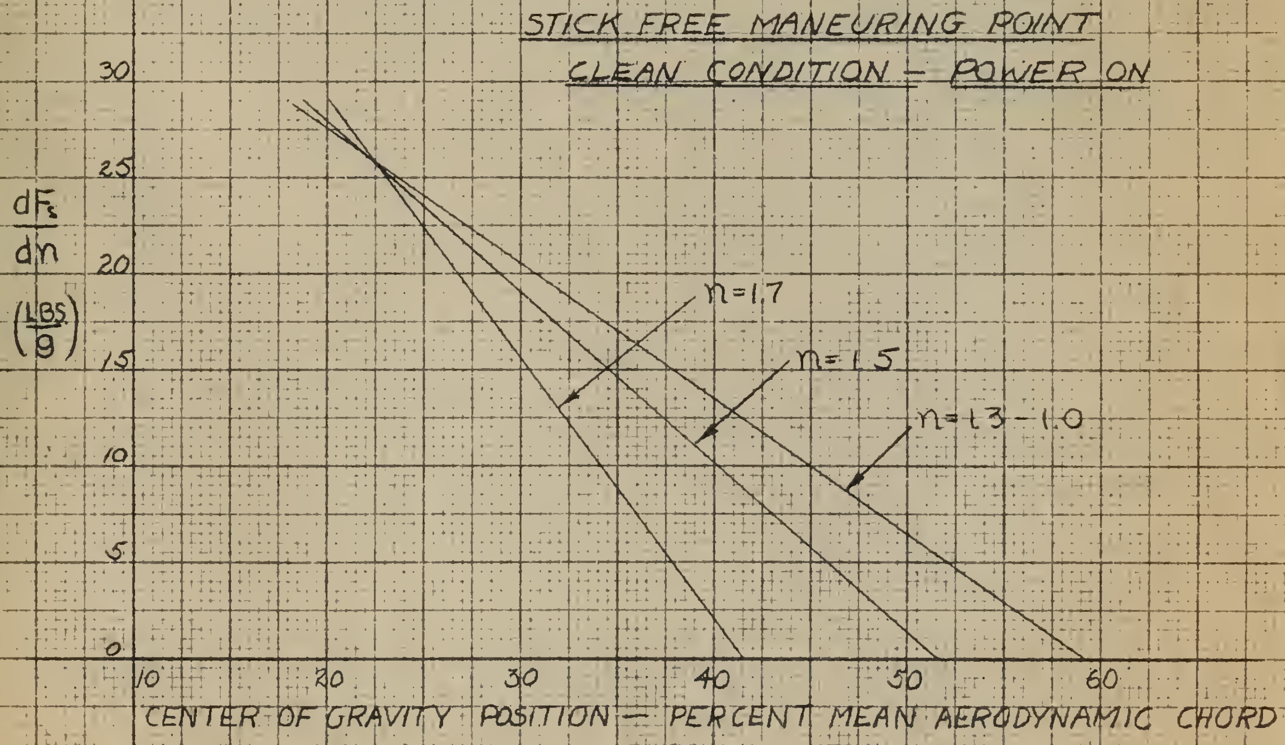
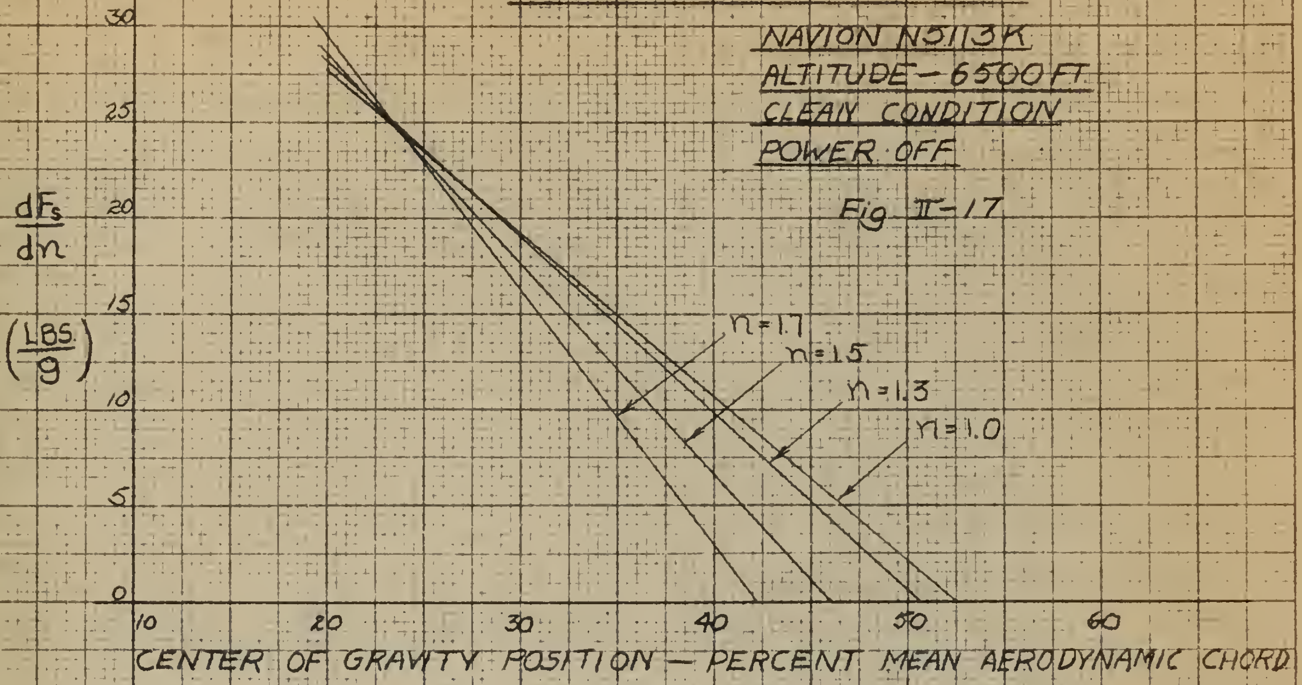

STICK FIXED MANEUVERING POINT
CLEAN CONDITION
NAVION - N5113K
ALTITUDE - 6500
$\frac{d\delta_e}{dn}$
$\left(\frac{\text{DEGREES}}{g}\right)$
7
6
5
4
3
2
1
0
10
20
30
40
50
60
POWER OFF
POWER ON
CENTER OF GRAVITY POSITION — PERCENT MEAN AERODYNAMIC CHORD

Fig. II-16


STICK FREE MANEURING POINT
CLEAN CONDITION - POWER ON
$\frac{dF_s}{dn}$
$\left(\frac{\text{LBS}}{g}\right)$
30
25
20
15
10
5
0
10
20
30
40
50
60
$n=1.7$
$n=1.5$
$n=1.3-1.0$
CENTER OF GRAVITY POSITION — PERCENT MEAN AERODYNAMIC CHORD



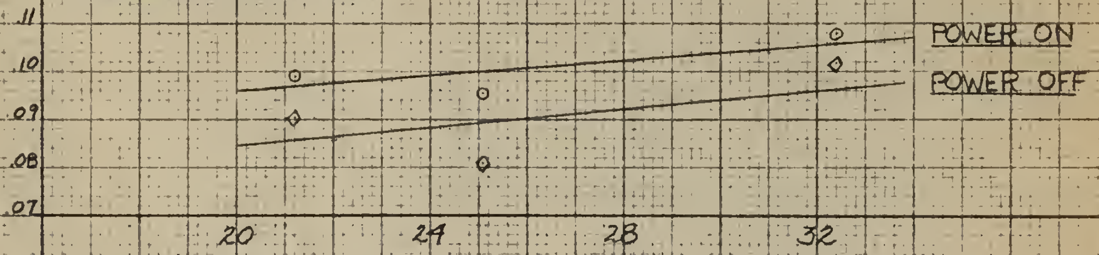
STICK FREE MANEUVERING POINT

NAVION N5113K
ALTITUDE - 6500 FT
CLEAN CONDITION
POWER OFF

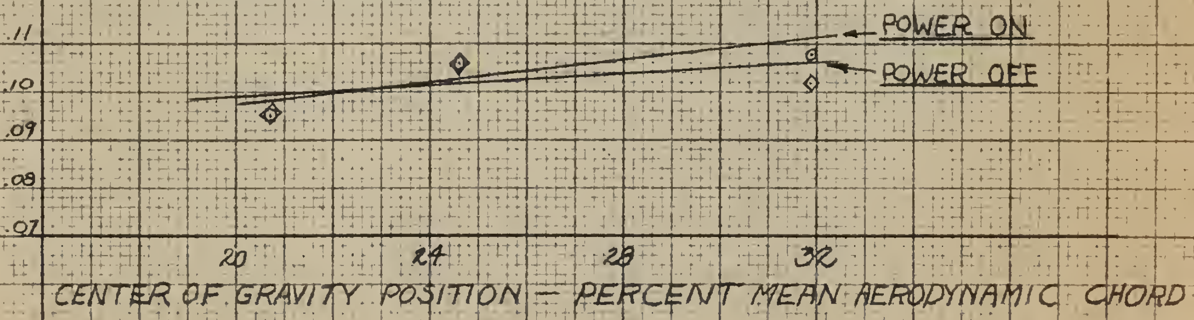
Fig II-17

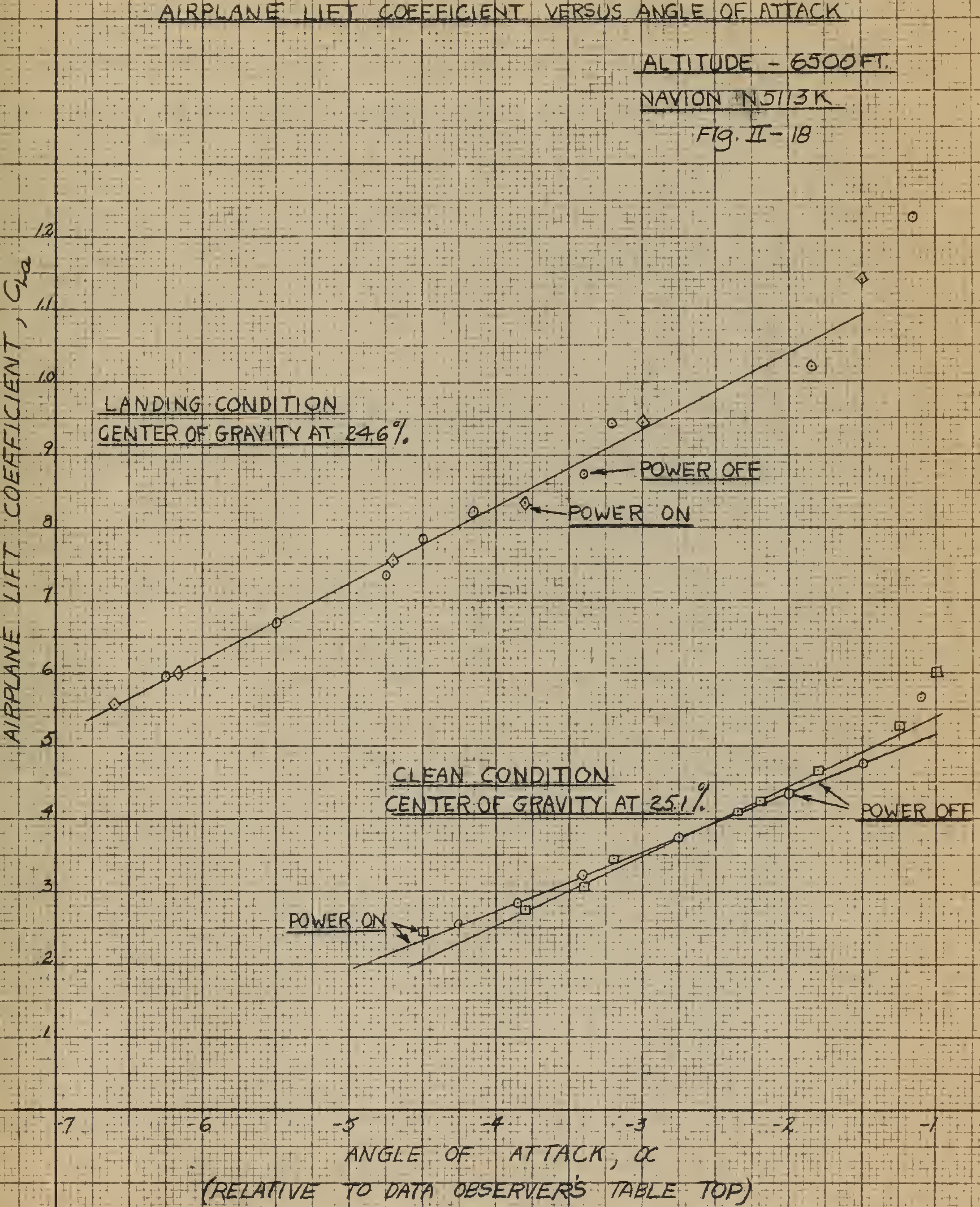
LIFT CURVE SLOPE VARIATION WITH CENTER OF GRAVITY MOVEMENT

CLEAN CONDITION



LANDING CONDITION



AIRPLANE LIFT COEFFICIENT VERSUS ANGLE OF ATTACK
ALTITUDE - 6500 FT.
NAVION N5113K
Fig. II - 18
LANDING CONDITION
CENTER OF GRAVITY AT 24.6%
POWER OFF
POWER ON
CLEAN CONDITION
CENTER OF GRAVITY AT 25.1%
POWER OFF
POWER ON
AIRPLANE LIFT COEFFICIENT, C_{L_a}
1.2
1.1
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
-7
-6
-5
-4
-3
-2
-1
ANGLE OF ATTACK, α
(RELATIVE TO DATA OBSERVER'S TABLE TOP)

APPENDIX A

The following is a tabulation of the results from Princeton Reports No. 231 and 232. Report No. 232 consists of the determination of longitudinal stability parameters, for a standard Ryan Navion, using theoretical calculations and steady state flight test techniques. Report No. 231 consists of determining as many as possible of these same parameters for the same airplane using dynamic flight testing techniques.

Clean - Power On

Parameter	Theoretical	Steady State	Dynamic
C_{L_α}		0.100	0.088
C_{m_δ}	-0.022	-0.023	-0.031
N_o	0.380	0.370	0.440
N_o'	0.420	0.440	
N_m	0.46	0.45	0.52
N_m'	0.53	0.59	
C_{m_α} (based on N_o)	-0.37	-0.39	
C_{m_α} (based on N_m)	-0.75		-0.67
$C_{m_{d\theta}}$	-0.17	-0.12	-0.18
$C_{m_{d\alpha}}$	-0.08	-0.06	-0.09

Clean - Power Off

Parameter	Theoretical	Steady State
C_{L_α}	.084	.090
C_{m_δ}	-.020	-.022
$C_{m_{d\alpha}}$	-.073	-.068
$C_{m_{d\theta}}$	-.15	-.13
N_o	.39	.39
N_o'	.42	.44
N_m	.46	.47
N_m'	.52	.51

Landing - Power On

Parameter	Theoretical	Steady State
C_{m_δ}	-.028	-.029
N_o	.39	.38
N_o'	.41	.43

Landing - Power Off

Parameter	Theoretical	Steady State
C_{m_δ}	-.020	-.024
N_o	.48	.41
N_o'	.50	.50

Examination of the tables above shows the following:

1. Values of C_{L_α} determined by the different methods were all within 12%.

2. Theoretical and steady state values of C_{m_δ} agree very closely while the value of C_{m_δ} from dynamic testing seems to be about 30% high. Some of this could be accounted for by the fact that non-linearities in the elevator angle potentiometer used for dynamic work caused a possibility of 10-15% error in idividual values of the elevator angle. However, it is improbable that the slope of the elevator angle calibration curve could be more than 10% off. In steady state testing, power for level flight was used. In the dynamic testing, less power was used in order to keep the engine RPM low to avoid angle of attack boom resonance. The 110 MPH air speed was attained by establishing a steady rate of descent.

3. The values of the stick fixed maneuver point $N_{m_{V=K}}$ again agree between theoretical and steady state values while the value from dynamic testing seems to be 17% high. The difference seems excessive since it represents a 47% descrepancy in the maneuver margin. The graphical plots of Figs. 3 and 4 of Report No. 231 indicate the difficulty of obtaining from dynamic data the intercept which determine the maneuver margin. Even though the intercept is defined by lines crossing at a small angle this can not account for the 47% discrepancy. Dynamic work was severely hampered by failure of the rate gyro and continued investigation with accurate

rate gyro measurements may yield better results. The effect of carrying the lower power for the dynamic testing is also apparent when the power-off maneuver margin from steady state results is considered. If the values from dynamic testing are compared with the power-off values of steady state results the discrepancy is 27% indicating that the effect of carrying the lower power in dynamic testing may be very large.

The value of C_{m_α} obtained from steady state testing is much lower than that from dynamic testing for the same reasons causing the discrepancy in the maneuver margin. The difference in C_{m_α} is further accentuated by the small value of the damping derivatives obtained from steady state testing. The theoretical value of C_{m_α} determined from the stick fixed neutral point agrees with the value of C_{m_α} from steady state testing and the theoretical value of C_{m_α} determined from stick fixed maneuver point agrees faily well with the value of C_{m_α} determined from dynamic tests. This shows that dynamic test results are very closely related to maneuvering stability while the steady state results are more nearly related to static stability.

The values of the damping derivatives $C_{m_{d\theta}}$ and $C_{m_{d\alpha}}$ obtained from theory and dynamic testing agree closely while those obtained from steady state testing are too low. This can be primarily accounted for by the fact that the damping derivatives are very difficult to extract from the steady state test data and are presumed to be in error.

4. The greatest variance between the theoretical and steady state flight test results, with the exception of parameters previously discussed, was found to be in the stick free maneuver point for the power-on, clean condition, and in the stick fixed neutral point for the power-off, landing condition.

It is believed that the discrepancy in the stick free maneuver point may be attributed to the fact that at low normal accelerations the stick force required is low and erroneous indications of stick force are possible because of the friction in the control column and elevator system. Data reduction and analysis indicated no change in maneuvering point between normal accelerations of 1.0 to 1.3. However, if an interpolation is made between the stick free maneuver point for normal accelerations of 1.0 to 1.5, a value of maneuver point of approximately 55% mean aerodynamic chord is obtained which corresponds quite closely with the 53% mean aerodynamic chord obtained from theoretical considerations.

The stick fixed neutral point for the power-off, landing condition which was obtained from steady state tests corresponds to the trends indicated in the other power conditions and configurations. However, the theoretical analysis indicated considerably more stability. Assignment of the error involved can not be done with any degree of accuracy; however, it is believed the major portion of the difference can be accounted for in the evaluation of the factor $\left(1-\frac{d\epsilon}{d\alpha}-\frac{d\epsilon_p}{d\alpha}\right)$ in the theoretical analysis.

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OC 7 57
25 SEP 65

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